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Aeronautical and Maritime Research Laboratory

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ABSTRACT

Many of the fixed and removable panels on the RAAF F-111 aircraft are made up of bonded honeycomb sandwich panels. Experience with the RAAF fleet has shown that a serious problem exists with degradation and damage of these panels. A review of the literature was undertaken to gain an understanding of the extent of this problem. It was found that panels were subject to large areas of adhesive bond separation and corrosion damage. This damage was believed to be caused by the ingress of water in the panel through poor sealing at the edges or after repair of the panels. Moisture in the panel is also believed to cause adhesive degradation that may reduce the strength of the bonds in such panels. At the same time the literature was surveyed to determine the design load cases for such panels. This information was used to develop a simple finite element model of a bonded honeycomb sandwich panel. This model was in turn used to generate data on the loading and failure of such panels. In addition, an understanding of current repair techniques was gained by review of the F-111 Structural Repair Manuals and the RAAF Engineering Standard C5033.

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Executive Summary

Many RAAF aircraft contain bonded sandwich panels that are made up of thin-face sheets, metallic or composite, bonded to aluminium honeycomb core. A serious problem encountered by such panels is the susceptibility of the panels to damage and degradation, particularly long-term degradation by moisture in the adhesive bonds used to consolidate the sandwich panels. A number of recent in-flight failures of bonded sandwich panels on both Royal Australian Air Force (RAAF) and United States Navy aircraft together with ongoing experience of panel failures during repair are examples of the type of problems that can occur with sandwich panels.

In order to manage this problem, it is necessary for the RAAF to be able to identify which components are most critical, develop an understanding of the critical failure modes of honeycomb components and determine the level of degradation that can be tolerated by these components. Also, it is important that current repair techniques are evaluated to quantify their effectiveness and if required new repair strategies based on current materials and technologies developed for bonded sandwich panels.

The work reported here was aimed at understanding the design, loading and failure of bonded sandwich panels and the extent of the degradation problem in the RAAF General Dynamics F-111 aircraft. To simplify the task the work was focused on representative structure on the F-111. Also, a simple numerical model of a typical F-111 panel was used to develop a better understanding of the loading and stresses in bonded sandwich panels.

In addition, a survey of typical RAAF F-111 defects and repairs was carried out. An understanding of the types of degradation, their location and size was gained from this work. Finally, the F-111 Structural Repair Manual was reviewed and compared to repair procedures as outlined in the RAAF Engineering Standard C5033. The review showed that C5033 is more comprehensive when detailing the procedures for the design and application of bonded repairs.

This review has reinforced concerns that adhesive degradation is a serious structural integrity issue for sandwich panels and has highlighted a number of problems with damage limit specification and repair procedures for these panels. Also, it has given a better understanding of the issues that lead to adhesive degradation and the critical load cases for panel failure.

Authors

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Sally graduated with a BSc (Hons) in Materials Science from Monash University in 1996. Since joining AED in 1998 she has conducted research in the area of bonded composite repairs. Projects Sally has worked on include the F/A-18 aileron hinge repair, and research into embedding Bragg grating optical fibres as health monitors within bonded composite repairs.

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Richard Bartholomeusz graduated from the Royal Melbourne Institute of Technology with a Bachelor of Aeronautical Engineering. On completion of his degree he joined the Royal Australian Navy as an Aircraft Enginneering Officer working on an operational helicopter squadron He has completed an Associate Diploma in Management and a Graduate Diploma in Aircraft Engineering Management. After joining AMRL, Richard has been invloved in the design, developemnt and testing of advanced composites and adhesively bonded joints. In particular, he has worked on the development of battle damage repair techniques and the manufacture and design of composite structures. His areas of interest are the use of composite materials and bonded joints in aircraft structural repair.

Contents

1. Introduction

Many Royal Australian Air Force (RAAF) aircraft contain bonded sandwich-panels that are made up of thin face sheets, metallic or composite, bonded to aluminium honeycomb core. A serious problem encountered by such panels is the susceptibility of the adhesive bonds used to consolidate the sandwich-panels to long-term degradation by moisture. A number of recent in-flight failures of bonded sandwich-panels on both RAAF and United States Navy aircraft together with ongoing experience of panel failures during repair has suggested that degradation of these panels may be a serious problem. Degradation can occur both at the adhesive bond between the face and the core as well as the node bonds within the honeycomb itself. Other forms of damage include corrosion and disbonds that may be also related to moisture leaking into the panels.

In order to manage this problem, it is necessary for the RAAF to be able to identify which components are most critical, develop an understanding of the critical failure modes of honeycomb components and determine and assess the level of degradation or damage that can be tolerated by these components. It is important that the significance of degradation and damage on the structural integrity of a bonded sandwich-panel and then on the airworthiness of the aircraft be understood. Also, it is important that current repair techniques are evaluated to quantify their effectiveness and if required new repair strategies based on current materials and technologies developed for bonded sandwich-panels.

The aim of this preliminary study is to gain an understanding of the design, loading and failure of bonded sandwich-panels and the extent of the degradation problem in RAAF General Dynamics F-111 aircraft. A literature review was carried out on bonded sandwich-panel design and loading particularly on the F-111 aircraft. In order to gain a better understanding of the behaviour of bonded panels under load, a panel representative of F-111 fuselage structure was numerically modelled. The model was constructed using the original data used by the Original Equipment Manufacturer (OEM) to design the panels for the F-111 [1] and was compared to the analytic methods used by the OEM.

In addition, a survey of typical RAAF failures and repairs was carried out. Again representative structure on the F-111 aircraft was chosen for this review due to the availability of the data. An understanding of the types of degradation, their location and size was gained from this work. Finally, the F-111 Structural Repair Manual (SRM) [2] was reviewed and compared to current repair procedures as outlined in the RAAF Engineering Standard C5033 [3] (C5033). The review showed that C5033 is more comprehensive when detailing the procedures for the design and application of bonded repairs.

2. Bonded Sandwich-panel Design

Sandwich structures consist of thin, high-density outer and inner facings, and a thick, low-density core. They overcome the problem of increasing weight with increasing material thickness and are thus particularly useful in aerospace applications [4]. In sandwich structures, the core sustains the shear load, while the faces take the compressive and tensile bending loads. The faces also resist the shear and normal loads applied in the plane of the fuselage skin [5]. The honeycomb panel structure has excellent resistance to sonic fatigue cracking, due to its high stiffness to weight ratio and these types of constructions offer superior insulating qualities and design versatility. [4, 6]

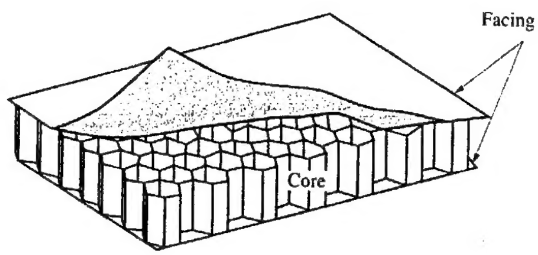


Figure 1. Honeycomb sandwich construction. [5]

The following design considerations for honeycomb sandwich-panels are taken from Reference 7. There are four basic design principles that should be observed when designing a sandwich-panel, as follows:

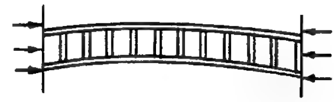
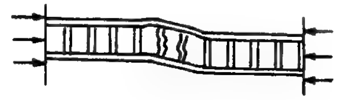
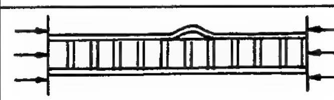
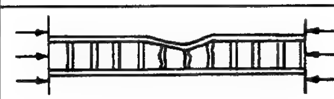
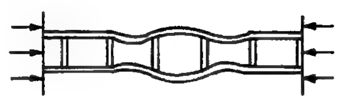
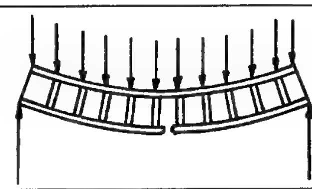
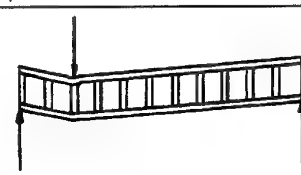
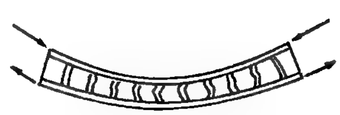
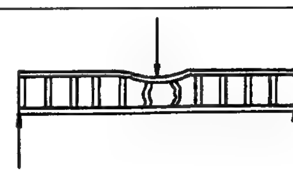
1. The facings need to be thick enough to withstand the chosen design compressive stress.
2. The core needs to be sufficiently thick and with a sufficiently high shear modulus so that buckling of the sandwich-panel will not occur at the design compression load.
3. The modulus of elasticity of the core needs to be sufficiently high and the flatwise tensile and compressive strength of the panel needs to be sufficiently high as to prevent wrinkling of either facing.
4. The honeycomb cell size needs to be sufficiently small so that dimpling of either facing into the core cell will not occur.

Also, the terminating edge of the sandwich construction is designed such that it has sufficient strength and stiffness to withstand the applied edge loads, and distribute those edge loads into the sandwich construction as uniformly as practicable.

2.1 Sandwich Panel Failure Modes

A number of buckling failure modes can occur depending on the relative strength and stiffness of the face, core and adhesive strength. Reference 8 gives a very good summary, repeated in Table 1, of the failure modes of metallic honeycomb sandwich-panels.

Table 1: Buckling modes of honeycomb sandwich-panels [8].

Buckling mode	*Cause*	*Mode shape*
General buckling (instability)	Insufficient bending stiffness, or Insufficient core shear rigidity	
Shear crimpling	Low core shear modulus, or Insufficient adhesive shear strength	
Face wrinkling	Thin face, and Low adhesive strength	
Face wrinkling	Thin face, and Low core strength	
Intra-cell buckling or dimpling	Very thin face, and Large core cell size	
Face failure	Lateral pressure Low face strength	
Core shear buckling	Insufficient core shear strength	
General core compression buckling	Insufficient compressive core strength	
Concentrated core compression buckling	Insufficient compressive core strength	

2.1.1 Adhesive Bond Failure

Adhesive bond failure is generally exhibited in three types of failure; face to core disbonding, core fillet bond failure and core node bond failure. These are described in Figure 2 (adapted from Reference 9). Degradation of the bond strength through exposure to moisture and free water will decrease the strength of these bonds. Sandwich panels are generally designed so that the core fails in shear before the face-to-core bond or fillet bond fails. Reduction in bond strength may change the failure mode and cause premature failure of the panel.

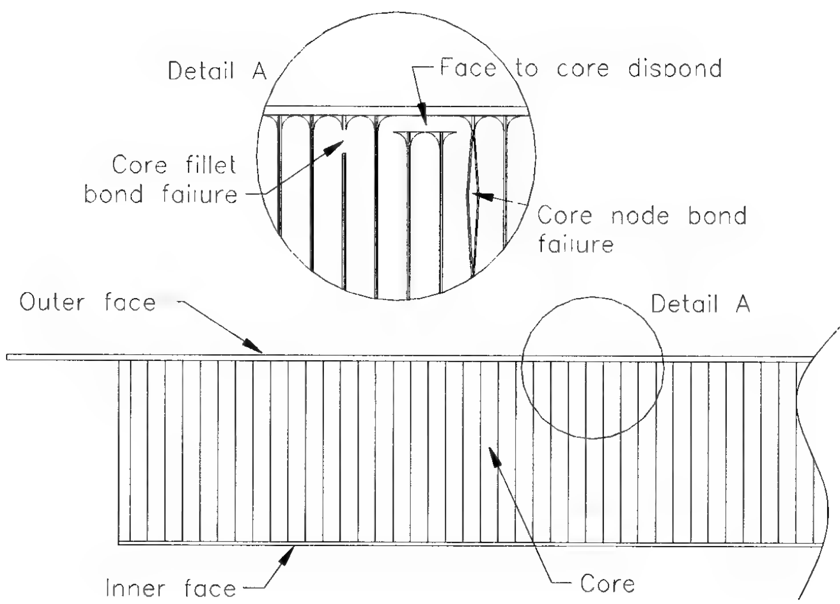


Figure 2: Failure of adhesive bonds in sandwich panels after degradation.

2.2 Correlation of Degradation, Design and Failure

Correctly designed sandwich-panels will resist the types of failures seen in service. Typically the in-flight failures take the form of the face or panel skin separating from the core. During repair, failures generally occur during panel heating when internal pressure exceeds the degraded face to core bond strength. This situation can occur when moisture trapped in the core boils and bond strength is reduced at elevated temperature [10].

Adhesive degradation may affect the face-to-core bond, fillet bonds and the core node bonds. Degradation of the face-to-core bond or fillet bonds may lead to shear crimpling and face wrinkling of the panel. Degradation of the node bonds may effect the core shear modulus and thus effect the shear buckling, crimpling and face wrinkling resistance of sandwich panels. Also, it may reduce the general buckling

stability of the panels. Degradation in the adhesive bond would also reduce the flatwise-tension strength of the face-to-core bond.

It is not surprising that adhesive degradation would lead to the types of failures seen in flight and during maintenance in sandwich-panels. Reduction in the face-to-core or fillet bond strength, particularly in combination with some form of void or disbond, may cause wrinkling or crimpling of the panel which in combination with low core-shear strength and panel-buckling instability may lead to separation of the face from the core.

3. Review of the design of the F-111 bonded sandwich-panels and preliminary FE analysis results

The above review of the design principles, loading conditions and failure modes of honeycomb panels was undertaken to better understand the conditions which lead to high adhesive stresses, and the potential consequences of adhesive failure. In an attempt to further understand the behaviour, loading and failure modes of such panels a numerical analysis of a representative panel was undertaken.

3.1 F-111 Bonded Sandwich-panels

The General Dynamics F-111 aircraft in service with the RAAF has many bonded sandwich-panels that make up its fuselage skins and some of its aerodynamic devices. These panels have aluminium alloy faces that are bonded to either aluminium alloy or phenolic Nomex honeycomb core using medium temperature curing film adhesives.

A major problem that was associated with these panels [11] was the degradation of the adhesive bond between the core and the faces, the bonds between the nodes of the core and general corrosion damage of the core and faces. The degradation or damage in these panels was attributed to poor sealing or repair procedures allowing the ingress of water. As such it was decided to examine the F-111 bonded panels as a case study that would be representative of the in-service condition of such bonded sandwich-panels.

The F-111 panels consist of a detail edge member design, shown in Figure 3. The outer face thickness is stepped to transition the perimeter loads from the outer face into the sandwich construction. The efficiency of this design, particular to the F-111, was not analysed in detail here. Also, the local failure modes are not well understood. In-service failures typically consist of the core disbonding from the edge member or the face. The consequences of the local edge-member bond failure were also not examined in detail. However, the internal load distribution in the core near the panel edge was investigated by using a simple finite element analysis (FEA).

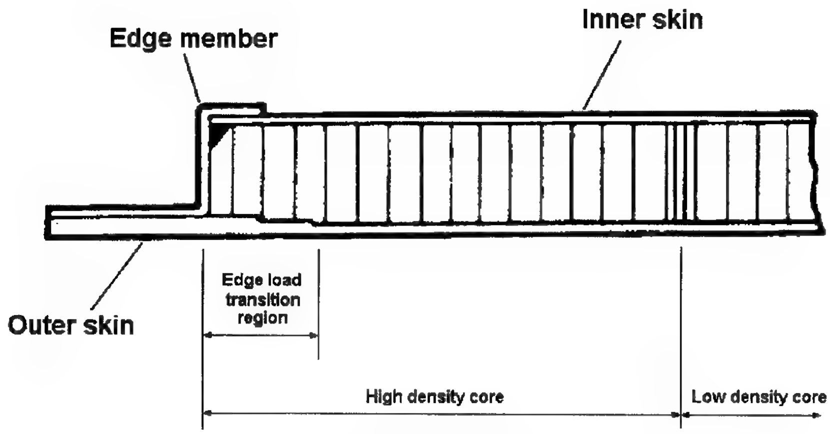


Figure 3: Typical edge member design for F-111 primary panels

Two panels were chosen on the F-111 for investigation:

1. Secondary panel - 1102 the forward equipment bay door.
2. Primary panel - 3208 the outboard nacelle panel.

The F-111 outboard nacelle panel was investigated in detail including a review of the OEM design calculations. A simple FE model has been used to provide some insight into the internal loading between the face and core. This panel is a primary load-bearing panel that has commonly seen the type of damage described above. Defect data was collected on both panels to determine the extent of the problem and the size and location of the typical damage in the fleet. This work is reported later in Section 4.

3.2 F-111 outboard nacelle panel design and loading

A review of the original design calculations of a primary honeycomb panel (Panel 3208, outboard nacelle panel) on the F-111 was conducted. Panel 3208 (Part Number 12B3913) is the forward most outboard panel on the aft centre fuselage side assembly (Figure 4). There are three such outboard panels on each assembly.

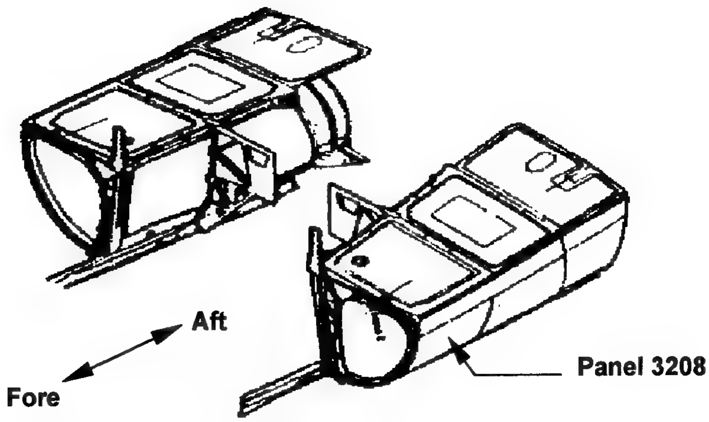


Figure 4 Aft centre fuselage side structures

The purpose of these panels is to provide shear support to the fuselage structure, as well as to provide an aerodynamic load path. In addition, the panels must withstand all other internal loads caused by bending, shear and torsion of the assembly.

3.2.1 Relationship between fuselage and sandwich panel loads

The primary loads in the panel are shear, axial bending and normal pressure loads. Panel shear loads are mainly caused by fuselage torsion and fuselage shear. Torsion in the fuselage arises during aircraft manoeuvres. Fuselage bending and shear are caused by the balance between the distributed fuselage mass reacted by the wing aerodynamic load. The fuselage bending component causes in-plane compression and tension loads in the panel while the fuselage shear component causes in-plane shear in the panel. Panel normal loads are caused by the aerodynamic pressure difference between the inner and outer surfaces of the panels.

The loads are introduced from the fuselage frame assembly into the sandwich construction via the outer face. The inner and outer faces are intended to act in combination to withstand these applied loads. Neither the outer or inner face can individually withstand the ultimate load. Panel loads are distributed from the outer face to the inner face via the edge member and the honeycomb core.

3.2.2 Relationship between panel loads and internal component loads

A panel shear load (in plane with the panel) induces an in-plane shear load in the outer face. This load is distributed partially into a similar in-plane shear load in the inner face, via an out-of-plane (the plane of this vertical shear load lies parallel to the edge of the panel) shear load in the core and edge member (see Figure 5).

The distributed panel normal load induces out-of -plane bending of the panel, and an internal shear load in the core which is, theoretically, zero at the centre of the panel and increases to a maximum at the edge of the panel (see Figure 6). The plane of this

vertical shear load lies parallel to the panel edge. This shear is transferred entirely to the outer face at the edge of the panel, via the core and the edge member (Note how the outer face increases in thickness toward the panel edge (Figure 3), this is to attract this vertical shear load into the outer face).

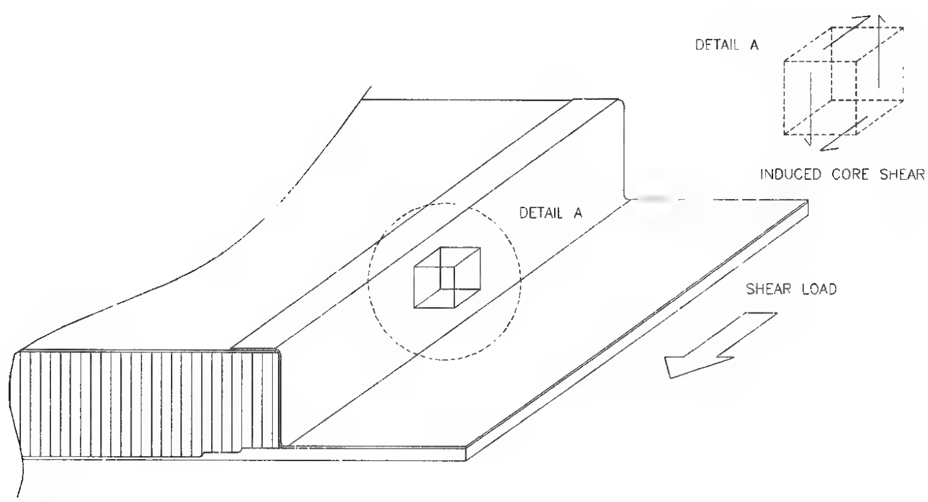


Figure 5: Torsion and fuselage shear lead to shear in the facings and an induced shear load in the core.

Compressive and tensile panel loads (in plane with the panel) induce compression or tension of the outer face. The core and edge members distribute this partially into a load of similar sign in the inner face. This distribution does cause some internal shear loading of the core (see Figure 7), and a more complex internal loading of the edge member.

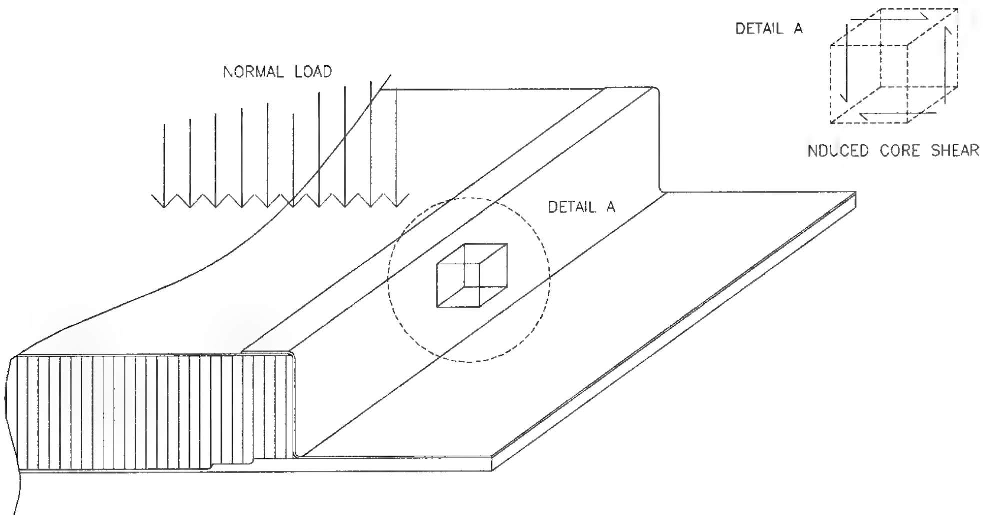


Figure 6: The normal pressure load causes out-of-plane bending and induces a shear load in the core

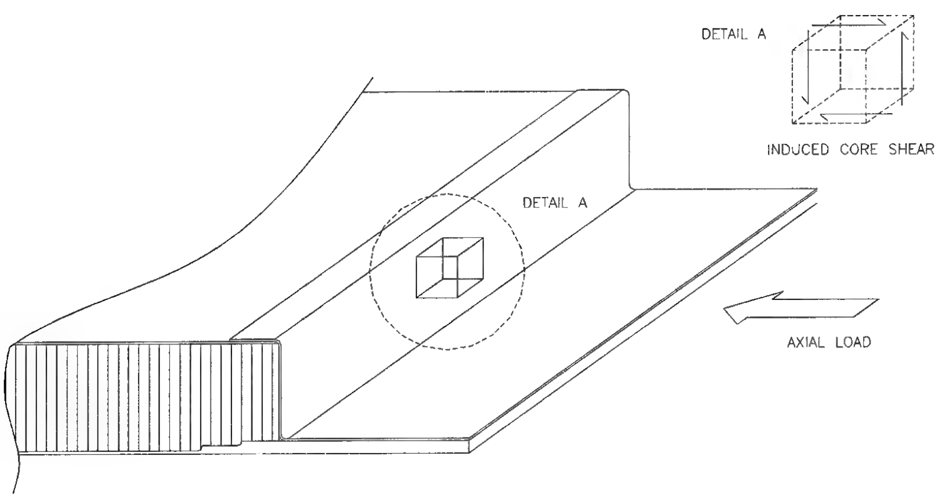


Figure 7: Fuselage bending leads to axial compression or tension in the face that induces shear in the core.

The internal loading in the assembly has a high degree of complexity and redundancy. For the original design these internal loads were determined using a computer math model [1] and the external pressure loads were generally hand calculated. In the original design process, the panels were subjected to two main internal load

components, a shear load and a compressive axial load. A schematic of the internal loads acting on the panel is shown in Figure 8.

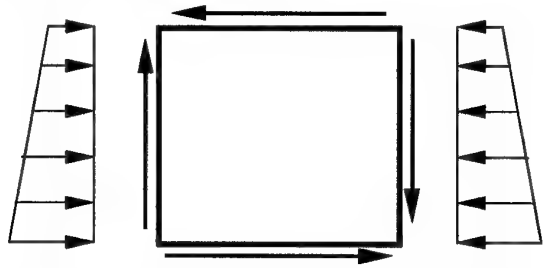


Figure 8 Schematic of internal loads acting on the outboard nacelle panel

It was noted that the panels were not permitted to carry axial load in the math model used to design the aircraft due to complexity. This is conservative for the design of the longerons, since the panels do provide some axial strain relief. For the panel design, a reduction factor was applied to the axial strains by comparing math model results to actual fuselage strain results under the same loading condition. It was considered that the reduced axial strains were more realistic for the panel design.

In the OEM design, each panel has its own unique set of loads for each flight condition. The most critical load components from all flight conditions were selected for the panel design. The panels were analysed using these critical load components both individually and in combination, even though they may not occur simultaneously in reality.

Several simplifications were made during the design of the panel. All edge loadings, shear and axial, were uniformly applied (as compared to the point loads at the fastener locations which occur in practice), and the calculations considered only a flat panel whereas in reality the panel has a very high degree of curvature. However, conservative selection of critical loads, stress calculation methods and allowable limits were intended to mitigate the complexity of the true stress field.

The panel design calculations consisted of a face analysis, core analysis and a fastener analysis. If the panel was relatively large, a stability analysis was also performed (this was not the case for panel 3208). The results were used to gauge the thickness of the aluminium faces and the core density. The basic design of the panel was not altered. The face analysis checked for overstress of the inner and outer faces. It was noted in the criteria that the panel faces were permitted to exceed yield strength under the ultimate loading. The core analysis checked the shear stresses in the honeycomb core near the panel edges. No considerable flatwise tension or compression stress was envisaged in the core, hence no attempt was made to estimate and check this. The fastener analysis simply consisted of checking the calculated fastener loads against allowable limits.

Although the individual metallic components were analysed with respect to externally applied loads (surrounding fuselage structure and aerodynamic pressure), it was noted that no detail explicit consideration was given to the internal loads between the core, face and edge members. The testing section of the honeycomb construction specification [12] required that the core must fail in shear prior to de-bonding. Hence, it is assumed that as long as the core shear stresses are less than the allowable core shear strength, then the adhesive stresses should also be less than allowable adhesive shear strength. This is reasonable for the face to core bond, however, no similar consideration was found for the edge member adhesive loads.

In summary, the face and core analysis of the F-111 primary panels generally ensure the panels have the appropriate stiffness and strength to overcome the buckling modes shown in Table 1. Given that it is known that in-service failures typically consist of face to core de-bonding and edge member de-bonding; the following were selected for further investigation by detail analysis:

1. Determine the load and stress distribution in the faces and core.
2. Determine the effect of gross degradation or damage in the panel.
3. Determine the effect of panel curvature.
4. Determine the effect of more realistic, non-uniform, edge loads.

Items (1) and (2) are addressed in this report and Items (3) and (4) are recommended for further investigation.

3.3 Simplified Finite Element Model

A simplified finite element model of panel 3208 was used to confirm the OEM calculations and provide some insight into the behaviour of the panel. Of particular interest was to determine if the flatwise tension stresses were significant and the internal load distribution near the panel edges. (Originally it was considered that degradation of flatwise tension strength of the face to core bond was the major contribution to in-flight panel failure.) The effect of de-bonded regions on the buckling strength of the panel is not considered in detail in this report, however, a simple extreme case of a panel with the faces completely de-bonded from the core was examined. The effect of panel curvature and more realistic load distributions were not considered in the analysis reported here.

Figure 9 shows the full three dimensional (3D) FE model of the flat, rectangular honeycomb sandwich-panel. The inner and outer face, and edge members were represented by plate elements. The honeycomb core was represented by 3D solid elements (Figure 10) assigned with orthotropic core material properties. The overall dimensions of the panel were representative of panel 3208, however the edge detail was modelled coarsely. Although the model did not aim to predict accurately stresses in the components near the edges, the general internal load distribution was captured satisfactorily.

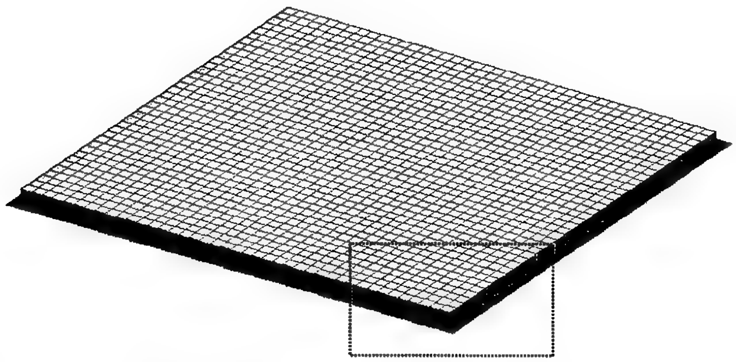


Figure 9: Element plot of simplified model of panel 3208

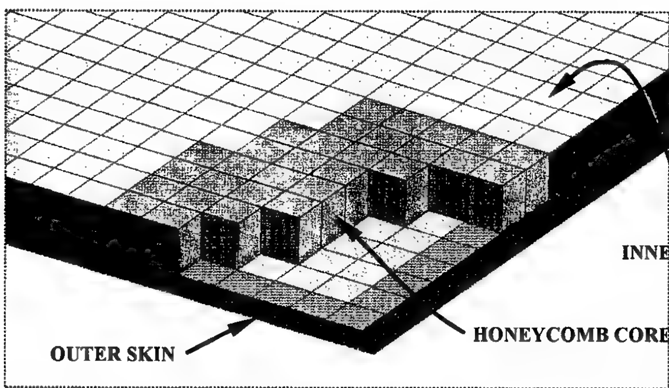


Figure 10: Detail view of elements showing sandwich construction of panel 3208

The OEM analysis implied that the stresses in the faces, under the ultimate combined load may exceed the yield strength. In reality, this may cause some load shedding from the panel to the surrounding structure, however, the OEM analysis conservatively assumes that this does not occur. It may also cause some internal panel load re-distribution, however this was not considered in the OEM analysis. For consistency and simplicity, the FE analysis also did not model load re-distribution due to yielding, hence it was performed using elastic material properties.

The results of the FE analysis indicate that the stresses in the centre of the panel for all load cases, individual and combined, agree very well with the OEM face analysis calculations. A stress contour plot of the panel under combined ultimate loads is shown in Figure 11.

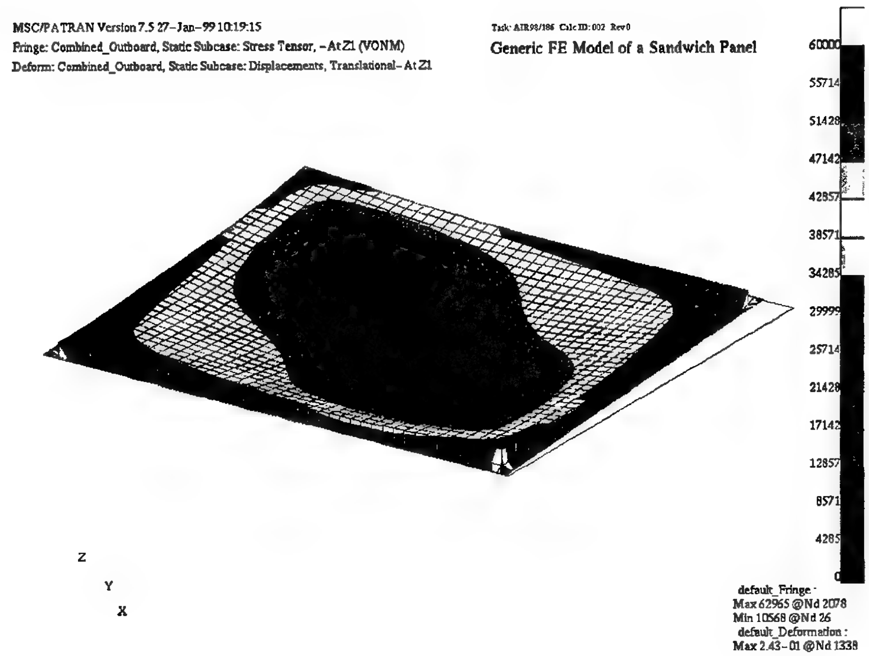


Figure 11: Contour plot of inner face stresses

A comparison of centre panel stresses from each individual load case indicated that the face stresses were dominated by the shear, which contributed about 70% to the total stress field, followed by the aerodynamic pressure, about 20%, and the axial load, about 10%. Note that the stress or load distribution here is typical of a fuselage panel and may be different for aerodynamic surfaces such as rudders, ailerons or flaps where aerodynamic pressure may play a larger role. Work has not been undertaken in this study to examine the load and stress distribution of such components.

Figure 12 shows a contour plot of the shear stresses in the core. The core shear stresses are representative of the shear loads experienced by the face to core bond. The stresses compare very well with the OEM calculations, despite the coarseness of the simplified FE model. (It is important to note that both the OEM calculation, and the FE analysis reported here, were both similarly simplistic and the true shear stress distribution in panel 3208 may be different.) The results also indicate higher core shear stresses near the panel edges (~250 psi).

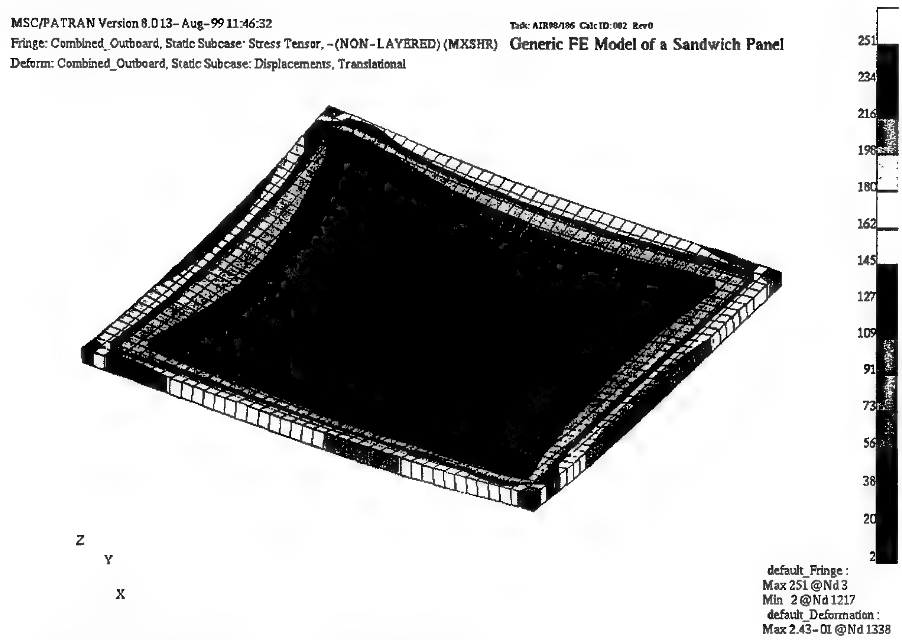


Figure 12: Contour plot of shear stresses in the core

The normal (flatwise tension) stresses in the core were found to be very small, and hence quite sensitive to the modelling approximations. In general, the normal stresses were an order of magnitude less than the shear stresses. This result was in agreement with the OEM analysis, which did not consider the normal stress effects in the core to be significant.

Closer examination of the internal load distribution in the core can be seen in Figure 13. This figure also shows how the shear stresses dominate the total core stress field.

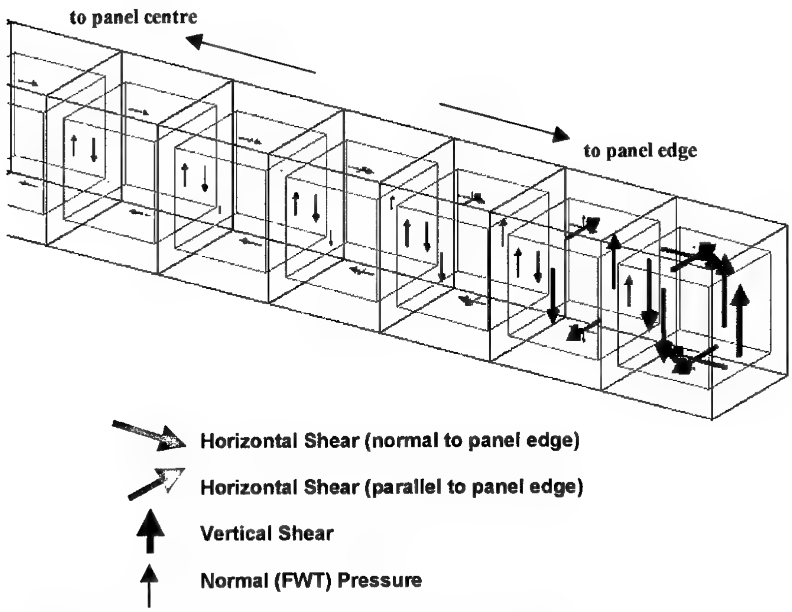


Figure 13: Internal loads in honeycomb core near the panel edge

Based on the findings of the simplified FE model, the application of the ultimate static load to the panel is sufficient to generate core shear stresses of around 250 psi. At this time no conclusions can be drawn regarding the ability of the panels to withstand the ultimate load in the degraded or damaged condition due to the lack of a knock down factor or allowable for panel degradation in sandwich-panels. Work is required to develop these knockdown factors.

3.4 Disbonded Panel

To determine the effect of a de-bonded region on the general stability of a panel, an extreme case was analysed where the core was totally disbonded from the faces. This was not a realistic scenario however it provided an upper bound to the panel failure problem. The simple FE model described above was analysed using a non-linear geometry option for both a completely intact panel and a completely disbonded panel.

The load-displacement response is shown in Figure 14 and this shows that the onset of general buckling instability begins at about 5% of ultimate load. Further, more realistic, work is planned which will consider the effect of smaller disbonded regions.

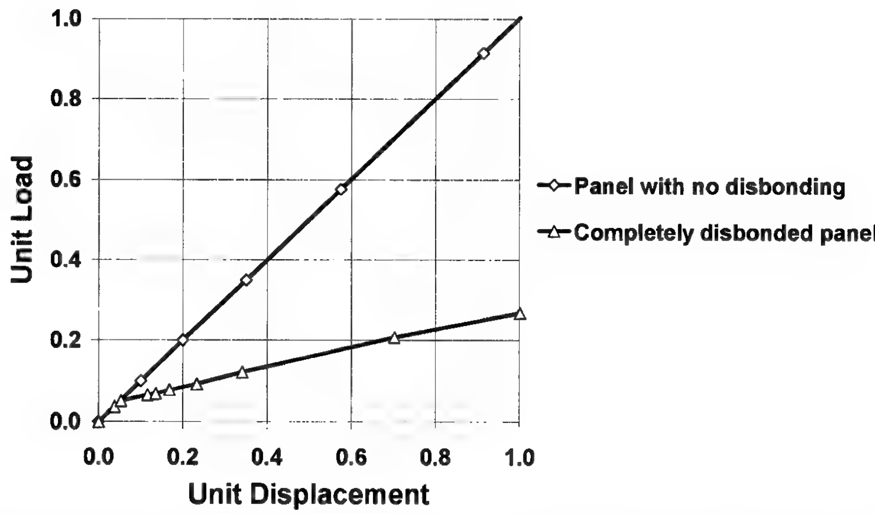


Figure 14: Load-displacement response of an intact panel and a completely disbonded panel.

3.5 Correlation of Analysis, Failure and Degradation.

Application of the design loads in the analysis showed that the stresses in sandwich-panels are dominated by shear, both in the faces and core. The hierarchy of failure in the OEM design was such that the core will fail prior to de-bonding or adhesive failure of the face-to-core bond. Thus it was assumed the core strength would be lower than the face-to-core bond strength, implying that adhesive degradation was not accounted for in the original design. As mentioned earlier, if the face-to-core or fillet bond degrades, the panel may fail in a mode not accounted for in the original design. Also, if a panel is completely disbonded, the analysis shows the failure strength of the panel reduces dramatically. Whilst not entirely realistic this shows that the panels are susceptible to failure if the face-to-core or fillet bond fails.

The analysis also showed that the highest shear stresses were at the edges or around the boundary of the panel. Figure 3 shows that the core density and stiffness around the boundary of the panel is higher than the core in the central region implying that this was accounted for in the original design.

4. RAAF Defect Review

We have gained an understanding that the strength of the face-to-core or fillet bond is one of the factors that may determine the integrity of the panel. To understand the implications of adhesive degradation on the RAAF aircraft, a review of service defect experience was carried out to identify the types of defects, failure and degradation that

are seen on the RAAF fleet. As mentioned earlier, to simplify the task two representative panels on the F-111 aircraft were chosen. The two panels described in Section 3.1, panels 1102 and 3208, were chosen for the review. Defect reports were obtained from the RAAF and while the information contained in the defect reports was not detailed enough to determine the nature of the failure or degradation, comments on the size and location of defects were made. Reference 11 is a comprehensive review of panel defects found in the RAAF that describes the nature of failures occurring in honeycomb sandwich-panels. The review described in this report spans defect reports from 1978 to July 1998.

4.1 RAAF Defect Reports

RAAF defect reports are raised for defects outside the SRM limits. They are the first step in obtaining an RFD/RFW[1], required to perform repairs not described in the SRM. Defect report information for this review was obtained from AFENG SRLMSQN 501 WG[2] [501 WG] at RAAF Amberley. Only a summary of each report was obtained that nominally contained a defect report number, corrective action file number, description of the defect and action taken.

The extent of the defect information for the two panels varied greatly between defect reports. Some reports contain very little or no information on the nature of the defect, whereas others describe the problem or damage area and the action taken to rectify it in detail. Hence, due to the variability of information and the size of the sample, no statistical inferences can be made. The reports do indicate the type and size of damage occurring in the panels.

4.2 Primary Panel

As mentioned earlier, the primary panel under consideration in this task was nacelle outboard fuselage skin panel 3208, Part Number 12B3913 (see Figure 15).

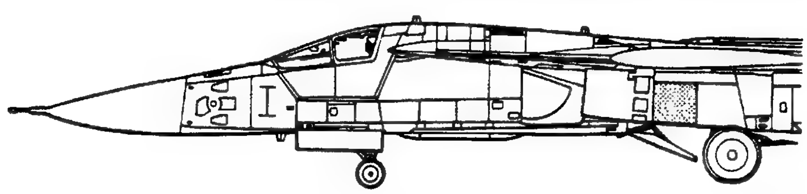


Figure 15: Location of panel 3208 (12B3913) on the F-111 aircraft (identified by shaded area) [2].

[1] Request for Disposition/Request for Waiver - repair authority issued when the damage exceeds the limits specified in the SRM

[2] Air Force Engineering Strike and Reconnaissance Logistics Management Squadron 501 Wing

The panel detail is shown in Figure 16. Although not obvious from Figure 16, the panel has considerable curvature. The detail of the construction of the panel including materials and face thickness is given in Appendix A.

4.2.1 Defect Details

An initial list of 14 defect report summaries were provided. Of these, five reports were deemed relevant and had further details available. Appendix B contains the numbers of the reports considered. Some reports relate to panel 3108, the panel with the same part number, but on the opposite side of the aircraft.

Table 2 is a description of the defects that have occurred in panel 3208 along with pictorial representations of the defects. (Note that these diagrams are not intended to be an exact replication of the defect, rather an approximate representation of their size and shape.)

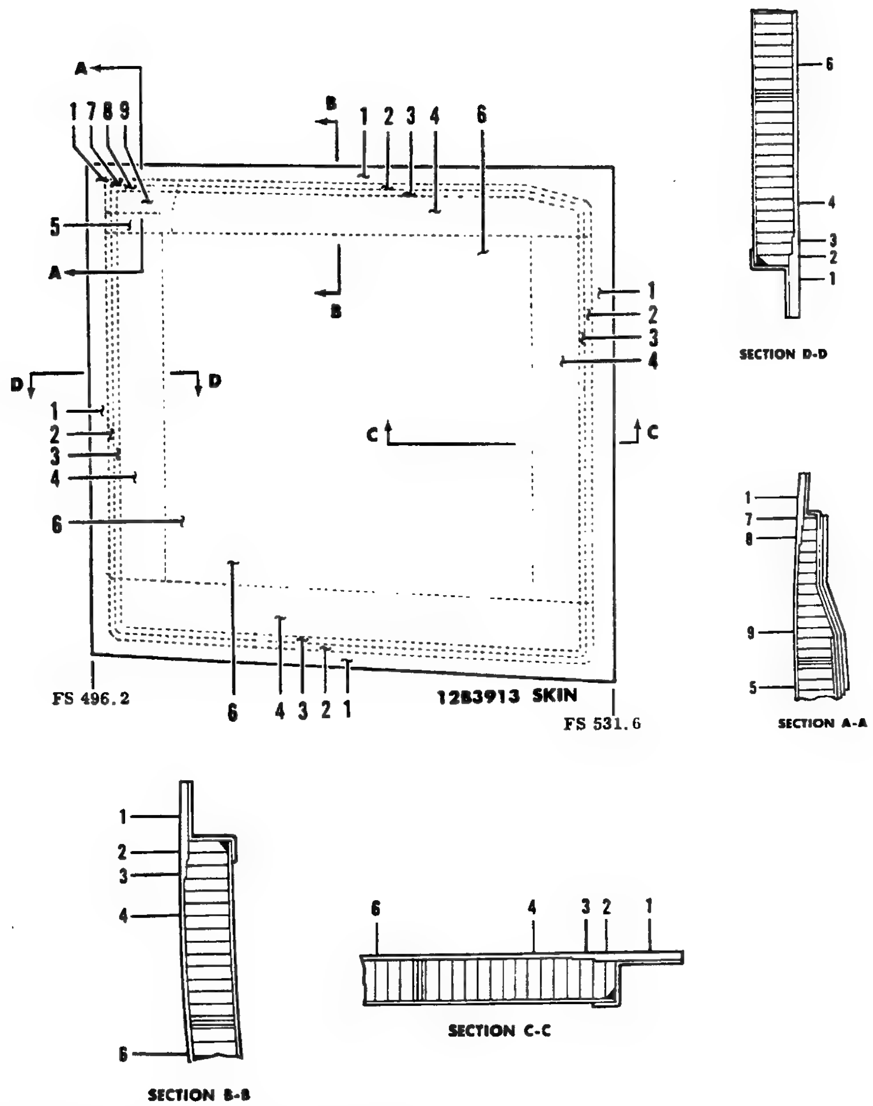
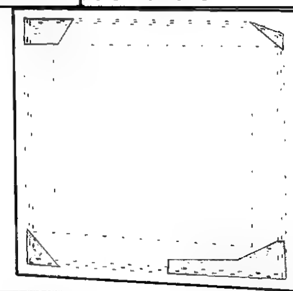
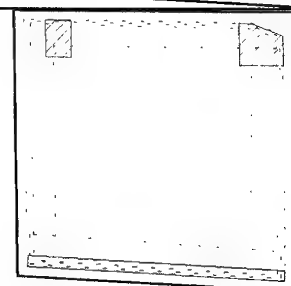
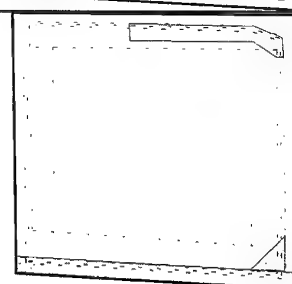
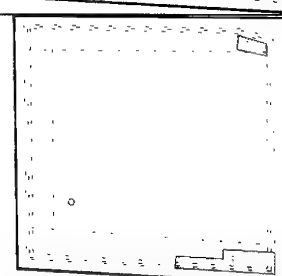
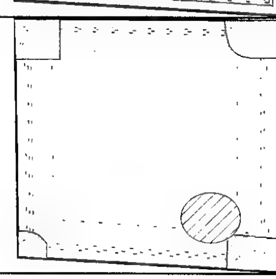


Figure 16: Structural detail of panel 3208. [2]

Table 2: Relevant defect information for panel 3208. (Cross reference with Appendix B.) The grey shaded areas show the defective area; the striped areas signify existing repairs.

	Nature of defect	Cause of defect	Size and shape of failure
A	Edge members on all four corners disbonded	Water ingress to panel	
B	Panel edge member cracked	-	
C	Outer face to core disbond at lower corners and along lower edge	-	
D	Core-face disbond along lower edge of panel	-	
E	Several disbonds on panel	-	

4.2.2 Evaluation of Defects

It was evident that the majority of the defects for panel 3208 occurred at the panel corners and along the panel edges. This pattern of disbond damage distribution was intuitively expected because the edge of the panel would be the most likely source of water ingress into the panel.

Some defect reports described quite extensive damage, in particular, defects B and C show the edge member disbonding along the entire length of the edge. The majority of reports describe multiple damage sites; for example defects A and E describe disbonding at each corner of the panel. Here the area of the corner disbonds was of the order of four to five square inches.

Some of the reports contained repair procedures and these were either oversize D12 or D14 (see Section 5.1.3), however, the majority of the reports did not list a specific repair procedure that was applied. In considering the integrity of repairs, note that one defect report (Report B) described a repaired edge member beginning to debond. This raised questions into the durability of the repair procedure.

The defect reports do not specify whether the disbonding occurred on the inner or outer faces of the panel. The descriptions of the defects were not necessarily consistent with the diagrams of the defects and standard descriptors are not used. It should be noted that the defect reports list "moisture ingression" as the cause of the defect.

There was insufficient information regarding the nature of the defect, for example, a description of failure surfaces, and thus the likely failure mechanism could not be ascertained. However, Davis [11] reported that operational stresses are not the cause of these types of failures. This was reasoned because the loads on honeycomb panels are low and would not usually generate adhesive failure. In addition, sandwich-panels are designed so that the core fails before the face to core bond, provided the bond strength is not degraded.

4.3 Secondary Panel

The secondary panel considered was panel 1102 (part number 12B11521), the equipment bay aft door, the location of which is shown in Figure 17.

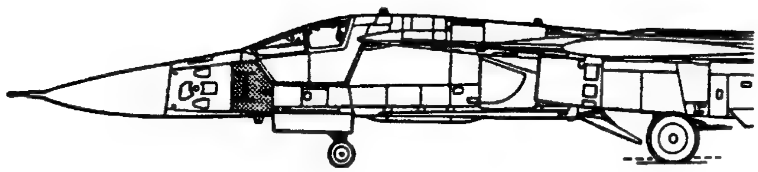


Figure 17. Location of equipment bay aft access door (identified by shaded area) [2].

Detail of the panel's front view and cross-section is given in Figure 18. The construction details are given in Appendix A. There is an antenna attachment in the centre of the panel.

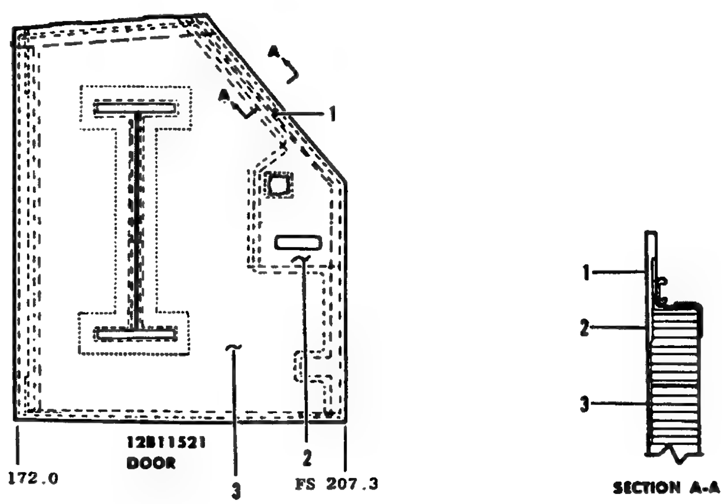


Figure 18. Panel 1102 structural detail. [2]

4.3.1 Defect Details

Initially defect data was extracted from the "Defect Summary Reports" on panel 12B11521-XXX and further information on the size and location of the defects was obtained from Reference 13.

Twenty-one defect report summaries were available. Five defect reports described electrical defects relating to the antennae or coaxial cable that were not relevant. Three defect reports had no details on the defect or the cause. Further information on the remaining thirteen defects was sought and of these, nine had further details available

[13]. Appendix B contains the full details of the defect report numbers and corrective action file numbers.

Table 3 contains information on the defects, as extracted from the defect report summaries. The diagrams show the location and approximate size of the defects on the panel.

4.3.2 Evaluation of Defects

Similarly to panel 3208, many of the defects on panel 1102 occur along or near the panel edges and corners. One defect (defect B) occurs near the antenna connection, which would also be a possible location for moisture ingress. There are two instances (defects E1 and G) of an entire edge member disbonding and similarly to panel 3208 some reports contained several defective areas (reports E, F and H).

While there are no reports of repairs failing after their application, two reports describe repair procedures exacerbating the defect size. In defect report C, the application of excessive heat caused damage to a large (20 x 10 inch) section of panel. In report H, the presence of moisture in the panel caused further disbonding of the panel during the cure cycle of the repair procedure.

For this panel, disbonding was reported between the core and both the inner and outer faces. The only repair procedure listed on some of the reports was oversize D14 (refer to Section 5.1.3.3).

4.4 Other RAAF Defect Report Reviews

The literature [11] describes a previous RAAF Defect Report survey for defects in honeycomb panels. This survey drew on a considerably larger number of defect reports (367). These reports did not include those associated with obvious impact damage. Adhesive bond failure was attributed to 53% of the defects with the most common causes being disbonding between core, face sheets and edge members.

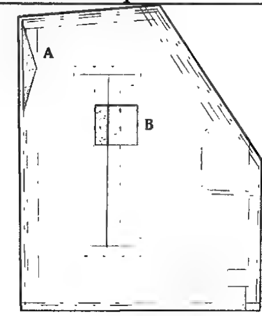
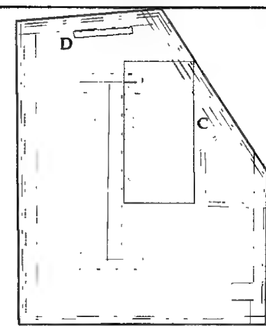
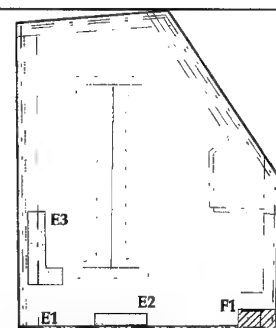
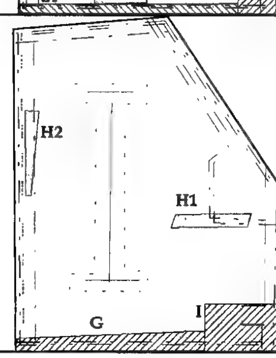
A large number (42%) of these defects described were associated with previous repairs. In considering the repair procedures, the literature in particular highlights the inadequacies of injection repairs where no surface preparation and poor heating procedures have caused failures at a later stage. These issues are considered further in Section 5.

4.5 Correlation of Defect, Failure and Analysis

It is believed that moisture or free water trapped in the core is the major cause of panel disbonding or corrosion in the defects reviewed above. The majority of the defects are found near locations that may allow entry of moisture into the panel, such as the edges

of panels and repairs due to poor sealing or sealant failure. More work is required to develop a better understanding of how moisture or water enters the panel.

Table 3: Detail of failure for relevant defects on panel 1102. (Cross reference with Appendix B.)

	Nature of defect	Cause of defect	Size and shape of failure
A	Edge member cracked and core delaminated	Over extension of door during opening	
B	Outer face cracked and	Moisture ingression	
C	Face buckled and disbonded	Application of excessive heat	
D	Face to core disbond and cracking along edge member	-	
E	Corrosion on external face	Moisture ingression through fastener holes	
F	Face disbonded	-	
G	Extensive corrosion on face	Ingress of moisture through door seal	
H	Face-to-core disbond, problem exacerbated during repair attempt	Moisture in panel; moisture caused further disbond during cure cycle of repair	
I	Face to core disbond	-	

It is of concern that the highest stresses are found along the panel edges increasing the significance of degradation or damage to the face-to-core or fillet bond in these locations. Also, it was noted that during some repair procedures the damage or defect size had increased (due to a build up of pressure in the panel during elevated temperature adhesive cure) and that complete edge member disbonds have occurred implying that the significance of edge member failure on the panel will need further investigation.

5. Review of Current Standard Repairs for Bonded Sandwich-panels

During repair, sandwich-panels have failed or defect sizes have increased due to trapped moisture in the panel boiling, increasing the vapour pressure in the core cells and blowing the faces off the core. To get a better understanding of this problem the current repair procedures for bonded sandwich-panels contained in the F-111 SRM and the RAAF ENG STD C5033 were reviewed. RAAF ENG STD C5033 is used for the management of composite and adhesive bonded repairs on ADF aircraft. The F-111C SRM is currently undergoing a major review with the intent of aligning the content with C5033. The intent of the review was to compare the two documents, look for inconsistencies and determine if the repair procedures were adequately defined. To simplify the task only repair procedures applicable to the defects reviewed above were considered.

5.1 Structural Repair Manual

This section describes the steps to determine the appropriate repair procedures as outlined in the F-111 SRM [2]. The review is restricted to the procedures relevant to panels 3208 and 1102.

5.1.1 Categorising the Repair Area

Several characteristics of the structure and damage need to be ascertained to determine the appropriate repair procedure for honeycomb sandwich structures. These are outlined in Appendix IV of the SRM and are summarised below,

1. Repair area.
2. Design temperature range of panel.
3. Face thickness, core type and core density.
4. Damage type.

The aircraft panels are characterised into four repair areas that are described in the SRM as A, B, C and D and are set by panel construction, the margin of safety on the original panel and the materials used in panel. Repair Area A is the most critical. For

panels 3208 and 1102, the repair area, the design temperature range of the panel, face thickness and core details are given in Table 4.

Table 4. Relevant properties for panels 3208 and 1102 [2].

Property	Panel 3208	Panel 1102
Repair area	A	B
Temperature range (°F)	> 350	< 350°F
Face thickness (inches)	outer: 0.032 to 0.120	outer: 0.016 to 0.110
	inner: 0.020	inner: 0.012
Core cell size (inches), density (lb/ft³) and foil thickness (mm):		
Zones 2, 3†	1/8, 8.1, 2.0	n/a
Zones 4, 5	1/8, 8.1, 2.0	-
Zone 6	1/8, 4.5, 1	-
Zone 7, 8, 9	1/8, 6.1, 1.5	-

†Refer to Figure 16 and Figure 18 for details of the Zones within each panel.

5.1.2 Damage Evaluation

Panel damage is categorised as follows:

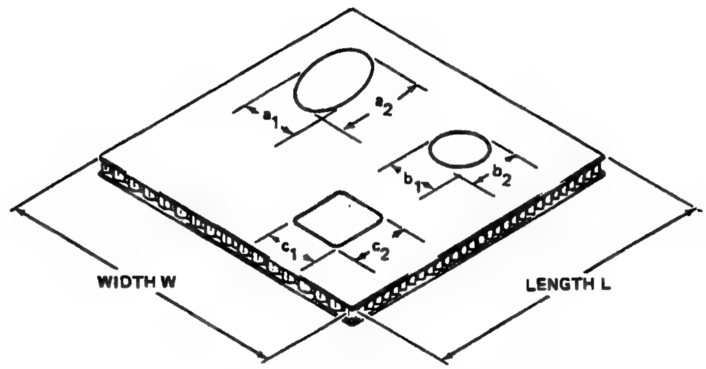
1. Scratches.
2. Dents.
3. Panel edge member damage.
4. Holes and cracks.
5. Disbonds and voids.

After considering the defect report information available (Section 4), the damage categories considered were "panel edge member damage" and "disbonds or voids". Table 5 describes both the dimensions of allowable damage and repairable damage limit for the above cases and details the preferred repair option as stated in the SRM. Components damaged above the repairable limits are subject to repair by engineering authority approval or are replaced.

Table 5. Allowable damage and repairable damage limit for panels 1102 and 3208 for edge member damage or voids and disbonds. Adapted from Figure D-5, pD-9 of App. IV in Reference 2.

			Panel 3208	**Panel 1102**	
Damage Type	**Allowable Damage Limit**		**Repairable Damage Limit**		**Preferred Repair**
Damage adjacent to panel edge member	Nil		3.0″ or less in width and 12″ or less in length	(as for panel 3208)	D-12
Damaged edge member	Nil		100% of bond length	(as for panel 3208)	D-25
Panel edge damage	Nil		0.500″ or less in width and 12″ or less in length	(as for panel 3208)	D-11
Face to core voids	Face thickness .011 - .020 .021 - .032 .033 - .051 .052 - .064 <.065	Allowable length 0.40 0.60 0.70 0.90 1.00	Less than 3.0″ in diameter or length	Less than 3.0″ in diameter or length. For face gauges <0.032″ and 4.0″ in diameter or length for face gauges >= 0.032″	D-18
			Void areas greater than 3.0″ but less than 12.0″ in diameter or length	Void areas greater than 3.0/4.0″″ but less than 12.0″ in diameter or length.	D-14

The appropriate repair procedure is determined from the Table 5, noting that only the preferred repair procedures are listed (the SRM also provided alternative repair procedures). The repairability is also dependent on any other existing repairs on the panel. A calculation that quantifies the "p-number", which is the percentage damage area of the panel, inclusive of any previously damaged areas (whether they have been repaired or not), is used to determine whether a panel is within repairable limits (as shown in Figure 19). The "p-number" allowable also depends on the type of repair that is applied.



$$p_1 = \left(\frac{a_1 + b_1 + c_1}{W} \right) 100 \qquad p_2 = \left(\frac{a_2 + b_2 + c_2}{W} \right) 100$$

p=p1 or p2, whichever is greater.

Figure 19. Calculation of percentage of panel width/length damage.

5.1.3 Repair Procedures

This section provides brief descriptions of the repair procedures appropriate for the damage commonly occurring on panels 3208 and 1102.

5.1.3.1 D11 Repair – Repair of Panel Edge Members

Edge members that are damaged on the faying surface of the panel faces are repaired by removal of the damaged outer face area, fabrication of a metallic plate to suit and application of the plate by riveting. Non Destructive Inspection (NDI) is used to identify the damaged area, which is removed by routing. A surface finish that comprises chemically treating and priming the repaired item followed by sealing all joints and faying edges is applied.

5.1.3.2 D12 Repair – Repair of Dent, Hole or Void in Face adjacent to Edge Member

This repair is for damage in the face or core adjacent to an edge member. The panel is cleaned and a moisture removal procedure carried out to dry the honeycomb core. The damaged area and edge member flange are routed to remove the damage. A core plug is manufactured and both the core plug and routed hole are prepared for bonding by performing a standard surface preparation for adhesive bonding. The plug and repair plate is then bonded to the damaged panel. An edge member angle cap is installed to cover the repair plate and edge member and finally, a honeycomb leak test carried out. A diagram of a finished repair is shown in Figure 20.

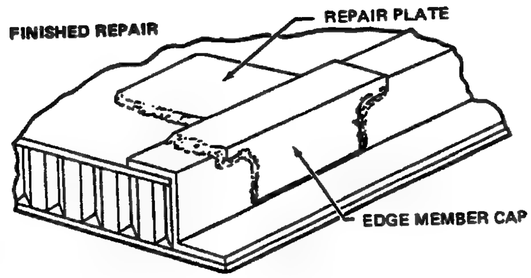


Figure 20. Finished repair for a hole or void in the face adjacent to an edge member.

5.1.3.3 D14 Repair – Repair of Hole using Overlap Patch

This repair is similar to D12 but without the complication of being adjacent to an edge member. The panel is cleaned, the damage removed and the panel is dried. The repair plates and core plug are then fabricated to suit. The surfaces to be bonded are prepared and the core plug installed, followed by the installation of the repair plate (note that the repair plate is non-flush) by bonding. Finally, the repair is surface finished and a honeycomb leak test performed.

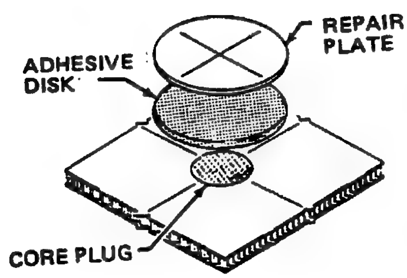


Figure 21. Hole repair using an overlap patch.

5.1.3.4 D18 Repair - Repair of voids

This procedure is prohibited for deeper level maintenance and is only accepted for operational level maintenance provided a patch is installed over the repair and that the repair be recorded as a temporary repair.

The procedure is used to repair voids in bonded panels between the face and the core, vertical edge members and core, core splices and metal to metal edges. The procedure involves the injection of epoxy potting compound into the voided region. The repair is essentially the same for each area, the only difference being the depth of the required injection holes.

As with other repair procedures, the panel is cleaned and the damage evaluated. With the aid of a template, holes are drilled in the panel face. The moisture removal

procedures are carried out to dry the panel and the potting compound is injected into the damaged area. The compound is injected into each hole until it is flowing from adjacent holes and then filled level with or above face contour. The holes are covered and thermocouples and a heater blanket are installed to cure the resin. After curing and cooling, the area is surface treated and a bonded cover pate installed. The repair is inspected and a leak test performed.

5.1.3.5 D25 Repair – Repair of Damaged Edge Members

This repair replaces damaged edge members. The damaged edge member is removed, the repair zone is dried and then prepared for bonding. A new edge member is manufactured and bonded to the structure. Following bonding the repair is inspected and a surface finish applied.

5.1.4 Notes on Repair Procedures

5.1.4.1 Moisture Removal

The moisture removal or drying process involves initially wiping the area clean with solvent. The zones to be dried are outlined and vent holes are drilled in the face to allow the moisture to evaporate. The area is heated to a maximum of 240 to 260°F using heater blankets and a vacuum bag (Reference 2 contains details of the heating cycle, including heat-up rates and temperature-holding times). The panel is re-inspected for moisture and the procedure repeated if moisture is still present.

5.1.4.2 Material Selection

5.1.4.2.1 Repair Plates

The general specification for repair materials is outlined below:

1. Internal riveted and bolted repairs - same material as damaged component one gauge thicker.
2. Patch repair filler plates - same material and gauge as damaged component.
3. Bonded external or internal repair - same material as damaged component, same gauge or greater.

This is consistent with the repair materials prescribed for each repair procedure described in this review, with the exception of repair D11 which specifies the use of aluminium alloy 2024-T86 instead of the face material which is 2024-T81. The repair material thickness requirements for each of the procedures are also discussed in Section 5.1.4.3.

5.1.4.2.2 Boron Fibre Crack Patching Repair Procedure

The SRM describes a "Boron Fibre Crack Patching Repair Procedure" for the repair of cracks in the structure. It is normally restricted to the repair of stress corrosion cracking and corrosion damage in aluminium structures and the procedure is only

used following a RFD/RFW. A similar procedure is described for "Graphite Fibre Crack Patching". These procedures are not expanded upon, and they are not referenced by the honeycomb sandwich-panel repair procedures.

5.1.4.2.3 Adhesives and Resins

The type of the adhesive system used (film, foam or liquid) needs to match the design requirements for the repair zone. The preferred adhesive for use with bonded repairs for the F-111 is the supported film adhesive FM300. The maximum service temperature of FM300 is 300°F hence a different adhesive is required for higher service temperature repairs. Other metal to metal and metal to honeycomb adhesives approved are FM73 (250°F), AF-130-2 (350°F), Plastilock 655 (400°F), EA9601 and AF-131 (450°F).

The honeycomb core splice adhesive films approved are Plastilock 654HE and Plastilock 653EX, that are both rated to a service temperature of 350°F. The approved potting compounds are Hysol EA 9317 NA A/B, EA934 NA A/B or EA9321NA. Each of these compounds has a wide service temperature range from -65°F to over 350°F.

5.1.4.2.4 Core Plug

Replacement core plugs should be made from the same material as the original honeycomb. Allowable substitutions of aluminium honeycomb core material are outlined. In the manufacturing of a replacement core plug, the plug needs to be made with a 0.050 inch mating clearance on all edges. It also needs to protrude 0.125 inches above the face. The core should be installed such that the ribbon direction is the same as the surrounding core.

5.1.4.3 Repair Plate Design

The design requirements of repair plates and edge members specify the necessary overlap lengths, dependent on the face thickness, and these are shown in Table 6. In addition, repair plate edge tapers are specified as shown in Figure 22.

Table 6 Overlap distance for repair plates and edge members.

Face thickness (inches)	Overlap distance (inches)
0.008 to 0.015	0.80
0.020	1.00
0.025	1.25
0.030	1.50
0.035	1.75
0.040	2.00
0.045	2.25
0.050	2.50
0.055	2.75
0.060 and over	3.00

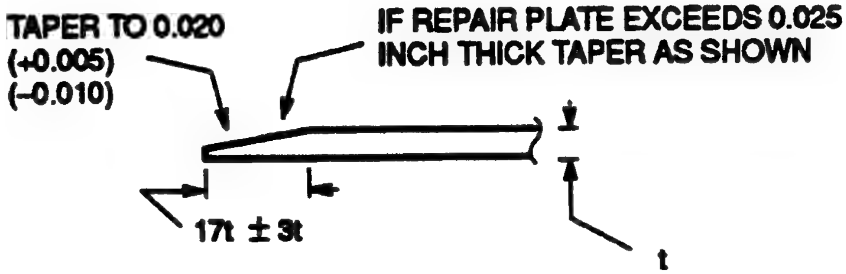


Figure 22. Taper dimensions for non-flush repair plates exceeding 0.025 inches thick.

The required repair plate thickness varies depending on the repair procedure, as follows:

D10 - 0.012 inch thick (cover plate).
D11 – repair plate is required to fit the cut out.
D12 – same thickness as the inner face, or the next highest gauge.
D14 – no thickness specified.
D15 – twice the thickness of the original material.
D18 – 0.012 inch thick (cover plate).

While the cover plates have a set thickness, the other external repair plates are of a thickness the same or greater than the original material. This is consistent with the general specifications for repair plates, as described earlier in Section 5.1.4.2.

5.1.4.4 Bonding Procedures

5.1.4.4.1 Surface Preparation

The SRM emphasises surface preparation as the most critical part of the adhesive bonding process. The SRM stresses that it is important to not to deviate from the procedures specified or an unsatisfactory bond will result. There are three stages in the surface treatment process for metals as follows:

1. Surface degreasing.
2. Exposing a chemically active surface, and
3. Chemically modifying this fresh surface

Several techniques are specified for the surface preparation of aluminium, as follows:

1. Tank Anodisation Method.
2. Non-tank Anodisation method.
3. Degrease, Grit-blast and coupling agent.

4. Degrease, Grit blast, Pasa-jell.
5. Degrease, wet-abrade with coupling agent, coupling agent.
6. Degrease, hand abrade, Pasa-jell.

The tank anodisation method involves vapour degreasing the component, cleaning the part by immersion in an alkaline solution, immersion in a de-oxidising solution followed by phosphoric acid anodising by immersion in a tank. The non-tank method involves degreasing and grit blasting of the surface to be bonded. A phosphoric acid gel is applied to the repair surface and contained by a fibre-glass frame. A fine stainless steel mesh grid is used as the cathode, the aluminium surface is the anode and the surface is then anodised. The tank method is only used for depot level maintenance and the non-tank method is also discouraged for general use. Technique 3 is the preferred technique for general repairs with Techniques 4, 5 and 6 to be used only for emergency purposes.

When effecting the surface preparation process, no more than 30 minutes is allowed to elapse between steps, if it does then the entire process must be repeated. The steps involved in Technique 3 follow:

1. Degreasing of the repair surface
2. Grit blasting, and
3. Application of the silane coupling agent to repair surface.

5.1.4.4.2 Surface treatment for honeycomb core

The preferred method for cleaning honeycomb core is by vapour degreasing. The alternate technique is to flush the core with solvent. Any sanding residue from pre-fit operations can be removed by spraying the component with an air/de-mineralised water mixture. The surface must then be adequately dried.

5.1.4.4.3 Heating

Moisture removal must be completed prior to undertaking heating for the cure of adhesives. Heating can be applied using an autoclave, furnace, heat lamps or heater blankets, as long as the cure cycle is appropriate to the adhesive, although heater blankets are normally used for the curing of bonded repairs.

When using heater blankets, the repair areas (and adjacent regions) need to be examined for substructural elements. The region is then divided into zones (according to thickness and material types) and a separate heater blanket is used for each zone. Heater blankets should not overlap - the dimensions of the heater blanket should match the dimensions of the zone. Other forms of heating, such as infra-red lamps, require special approval. Thermocouples need to be used in conjunction with heater blankets to ensure the correct cure cycle is applied to the adhesive. Thermocouples need to be installed on the aircraft, within each zone at the anticipated hottest point, and around the repair where the coldest temperature is anticipated to occur.

5.1.4.4.4 Pressure application

The SRM outlines a number of options for the application of pressure required for bonding:

1. Autoclave.
2. Vacuum bag.
3. Pressure box
4. Clamping,

The SRM specifically states that G-clamps are inadequate, as the pressure will decrease as the adhesive flows during the cure.

5.1.4.4.5 Post Repair Inspection

Most of the repair procedures described in this section require the completed repair to be inspected. Particular inspections referred to in the repair procedures include inspection for voids after a replacement core plug has been bonded into a panel cavity, (note that no voids are allowed in core splice areas). If found, the voids should be filled with adhesive. Metal to metal bonded areas need to be also inspected for voids and this is usually done using a "Tap Hammer Inspection" method. Some voids are allowed in bonded repairs and the acceptable void sizes are listed in Table 7.

Table 7. Permissible voids in bonded assemblies.

Face thickness	Max. allowable dimension in any direction
>0.010	0.00
0.011 to 0.016	0.025
0.017 to 0.020	0.038
0.021 to 0.025	0.050
0.026 to 0.032	0.055
0.033to 0.040	0.068
0.041 to 0.051	0.070
0.052 to 0.064	0.084
0.064 and greater	0.93

5.2 RAAF Engineering Standard C5033 [3]

5.2.1 Use of the Standard

The C5033 standard is to be used for the overall management of composite and adhesive bonded repairs on ADF aircraft. As authorised by the Engineering Authority, this standard applies to bonded and composite repairs, which exceed the limits specified in the SRM and to composite and adhesive bonded repairs developed as an alternative to the authorised repair manual. There are two associated publications, AAP 7021.016-1 and AAP 7021.016-2, which contain detailed explanation of the design

rules for repairs and process specifications for repair procedures respectively. The standard can be used as reference material in preparation for RFD/RFWs to the relevant SRM. The C5033 standard is not a repair manual for the F-111; the SRM takes precedence over C5033.

5.2.2 Damage Assessment

5.2.2.1 Defect Categorisation

As mentioned above, C5033 states that the defect assessment section must be read in conjunction with the SRM and that the SRM assessment takes precedence over C5033. C5033 contains a table of the type of manufacturing and service related defects that may be found in sandwich-panels. Comments on both the significance of defects and the basic repair procedures are made. These tables are included as Appendix C and give a holistic view of defects and damage occurring in sandwich-panels. A brief, broad repair strategy is included for each service defect type and it is of note that the repair strategy emphasises dealing with the cause of the defect rather than only dealing with the current manifestation of the problem.

5.2.2.2 Allowable Damage

C5033 refers to the acceptance criteria in MIL-A-83376A [14] for allowable damage limits in metal-faced sandwich-panels. The restriction on the size of one disbond or void area is described in Figure 23. It is implied in C5033 that these limits apply to the entire panel, that is the metal-to-metal bond area and the face-to-core bond area.

> An individual disbond must not have any dimension greater than 15 times the thickness of the thinnest adherend or 1 inch (whichever is smaller). They must not have dimension "S" greater than 15% of "W" as shown in Figure 23. Also, if W is less than 2 inches, S shall be less than 0.25 inches.

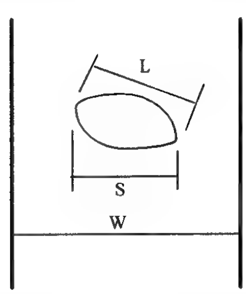


Figure 23. Explanation of symbols for permitted defect sizes in metallic sandwich-panels.

Figure 24 explains the minimum permissible spacings between voids as stated in C5033.

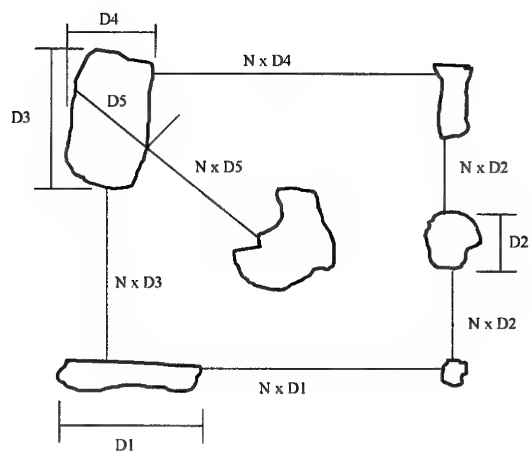


Figure 24. Minimum separation distances for multiple disbonds. For critical structures N=4 and for other structures N=3.

Further, no void is permitted within 0.125 inches of any edge of a metal to metal bond. Face to core disbonds must not exceed 0.5 inches diameter for critical structure and 0.75 inches diameter for other structure. In addition, there are restrictions on the dimensions of core splices and core-to-core bonds.

It is the authors' opinion that the implication of MIL-A-83376A is that the above limits only apply to the metal-to-metal bond area in a metal-faced sandwich-panel (C5033 suggests the limits are for the entire panel). Another section in MIL-A-83376A deals with limits for face-to-core bonds, stating "voids or disbonds shall be limited to 0.5 inch diameter for Type I structure and 0.75 inch diameter for Type II face-to-core or doubler-to-core bonds". No limits are suggested in MIL-A-83376A for the spacing between voids or disbonds in face-to-core bonds.

5.2.3 Repair Procedures

C5033 is a generic standard for bonded repairs and does not contain set procedures to follow for a defect in a given panel. C5033 describes generic steps to follow in designing and applying a repair. An overall view of repair criteria and design is described with emphasis on the repair satisfying a wide range of criteria, not just one or two factors. The particular procedures and processes discussed below are those that have been highlighted in the literature as often being incorrectly or poorly described in some repair manuals.

5.2.3.1 Moisture Removal

The drilling of vent holes to aid in removing moisture from honeycomb panels is not normally permitted. The reason for this is that vent holes increase subsequent fluid ingress in subsequent service operation. If the engineering authority requires that vent holes be drilled, a bonded patch must be applied to cover all vent holes.

5.2.3.2 *Material Selection*

C5033 emphasises the need for consistency of repair materials with the OEM. The engineering authority must approve the use of any material selection that is different to that prescribed by the SRM. C5033 does however state that the use of a particular adhesive during the production of an aircraft component does not necessarily mean that this adhesive is the best choice for the repair of the component. The production adhesive may be better suited to autoclave curing conditions and these conditions may not be achievable in the field.

Adhesives on the approved materials list are FM300, FM73, AF-130-2, EA 9321 and AF-131-2. Foaming adhesives approved for core splice and edge member bonds are FM-404 in addition to Plastilock 654HE and Plastilock 653 EX. There is no list of approved injectable potting compounds.

5.2.3.3 *Repair Plate Design*

5.2.3.3.1 Metallic Repair Plate

To determine the length and width of a bonded repair, the overlap length must be calculated. (Equations for determining the overlap length are given in the C5033 standard.) The overall patch dimensions are such that the patch must be extend the overlap distance in all directions. Metallic patches should be rectangular in shape (so the required overlap length can be achieved in all directions) and the corners should be rounded to a radius of at least 12mm. The thickness of the patch needs to be such that the patch stiffness is the same as (or up to 1.2 times) the parent material stiffness.

The requirement on tapering at the edges of a metallic patch is shown in Figure 25. There is also a requirement that the patch must be shaped to the contour of the panel. The patch needs to follow the surface profile to ensure the resulting adhesive film thickness will have a tolerance of +/-0.002 inches.

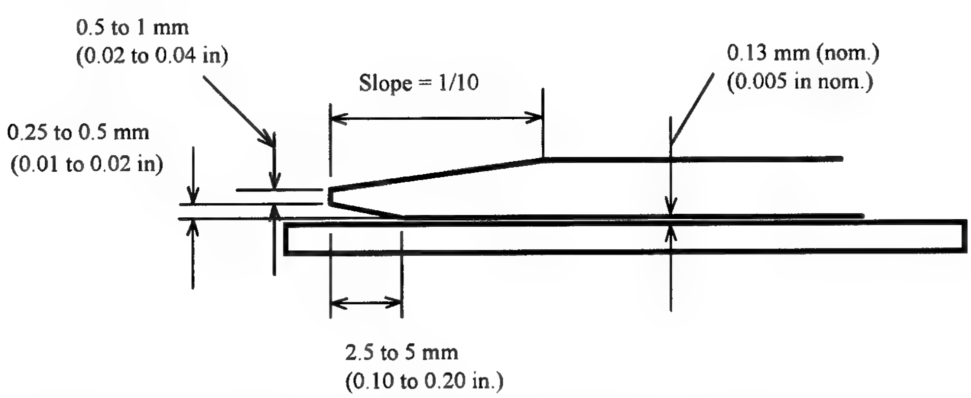


Figure 25. Tapering of metallic patches to reduce peel and shear stress at the edges of the repair.

5.2.3.3.2 Composite Repair Design

While the SRM only specifies metallic repair plates for the repair of damaged or defective structure (excluding stress corrosion and corrosion, which may be patched with a composite material), the C5033 standard allows composite repairs to be bonded onto metallic structures. While the composite repair design procedure is beyond the scope of this document, the repair methodology highlights that a number of factors (minimum patch thickness, thermal properties, weight, stiffness) need to be considered in determining the most suitable repair material (metallic or composite) for the repair. C5033 stresses the need to match the repair material to the facilities available and the expectations of the repair.

5.2.3.4 Bonding Procedure

5.2.3.4.1 Surface Preparation

C5033 highlights the importance of surface preparation to the durability of an adhesive bond. The three stages in the surface preparation process are degreasing the surface, exposing fresh material and then chemically modifying the material to suit bonding. It is noted that there is no post-treatment inspection available to verify that the surface procedures have been performed correctly.

To degrease the surface, solvent cleaning by vapour degreasing is the preferred method, however, when this is not available, hand solvent cleaning may be used. Methyl Ethyl Ketone (MEK) is the approved solvent, mainly due to its excellent degreasing properties and the fact that it very quickly evaporates from the surface during cleaning. Only Analytical Grade MEK is approved for the final stages of solvent cleaning and the use of detergents in lieu of a solvent is not approved. For exposing fresh material, hand abrasion should only be used when grit blasting of a surface is not possible. Chemical modification of the surface can be undertaken by the use of coupling agents or by acid etching. The preferred treatment is the use of silane-based organo-functional coupling agents. Acid etching processes are not encouraged as any residual acid on the metallic bond surface may cause corrosion at a later stage.

C5033 also specifies exposure times between each surface preparation step. The allowed exposure time is dependent on the atmosphere and is given in Table 8. In addition, if the surface is to be left exposed for more than five minutes, the surface should be covered by a piece of clean, new paper.

Table 8. Maximum surface exposure times for work environments.

Exposure condition	**Maximum Exposure Time**
Controlled area with air-conditioned, filtered air, and humidity control.	1 hour.
Controlled area, air conditioned with no humidity control. Hangar Floor inside a temporary controlled area with air conditioning and humidity control.	30 minutes.
Hangar Floor inside a temporary controlled area with air conditioning and no humidity control.	15 minutes.
Open Air	5 minutes.

5.2.3.4.2 Honeycomb Core Surface Treatment

The only surface preparation possible on honeycomb core is solvent degreasing and this can be achieved by immersion, flushing or vapour degreasing. Vapour degreasing is the preferred method as this avoids cross-contamination. As only a rudimentary surface preparation is possible for core there is an emphasis on careful storage and handling of honeycomb core.

5.2.3.4.3 Pressure and Heat Application

Due to the presence of moisture, there is a particular danger in damaging sandwich-panels during elevated temperature (<80°C) cure and they need to be thoroughly dried prior to repair. It is important to dry all areas on the panel that will be heated. This implies that if the cure is to take place in an oven or autoclave then the entire panel must be dried - not just the repair area. If only the repair zone is dried and the entire panel is heated, moisture from other areas within the panel may cause the face to debond.

Care must be taken when heating a component because incorrect heating procedures can either under-cure adhesives or cause damage to structures. There are two alternatives when heating a component for a bonded repair:

1. Heat the whole component - ovens or autoclaves are used for this. The oven must be a re-circulating oven, as radiant-heat ovens can cause very high surface temperatures and damage the part.
2. Heat only the repair zone - localised heating is more readily adaptable to in-field use however it can be difficult to achieve a uniform temperature distribution. Heater blankets are the most common forms of increasing cure temperature in aircraft repair situations. The use of infra-red heat lamps are not ideal as they can cause hot-spots and ceramic backed radiant heat lamps are prohibited.

The process specified to establish a suitable heater and thermocouple arrangement is essentially to divide the structure into separate zones and heat each zone with a separate heater blanket (or source) controlled by a thermocouple underneath the blanket. Several guidelines are provided to assist in establishing zones such that a uniform temperature distribution is achieved. These guidelines refer to the different heating rates of different materials and structural components. C5033 prohibits multi-zone heater blankets as the zones are circular in shape and are hence only suitable for concentric circular structures. There are also guidelines and limitations on the types of hot bonding units used for the control of temperature and pressure.

Thermocouples are recommended as a method of measuring the temperature during cure. Thermocouples need to be installed at the anticipated hottest and coldest locations on the structure. Thermocouples must be electrically insulated from the structure and thermally insulated from the heater blanket. Thermocouples contained within the heater blanket for control purposes are prohibited.

Several methods are available to pressurise an adhesive bonded repair including autoclave, vacuum bagging, clamping or dead weight. An autoclave is the preferred method but is generally difficult to use unless the part can be removed from the aircraft. In general vacuum bagging is the approved alternative.

5.2.3.4.4 Honeycomb Core Replacement

Core replacement with the use of foaming adhesives must be done with positive pressure, not under vacuum. Procedures involving bonding of core inserts and repair patches in a single process are not approved. The core insert requires 0.08 inch clearance around the cavity and should protrude 0.08 to 0.12 inches above the panel surface. For full-depth core replacements, if the adhesive layer is intact and has not degraded then there is no requirement to remove it. The standard pre-bonding surface preparation can be applied to the adhesive.

For partial core replacement, a thin metal barrier is placed on the exposed core surface to control adhesive flow during cure. The appropriate surface treatment is applied to the barrier and the core plug.

5.2.3.5 Post Repair Inspection

At the completion of a repair, the repair bondline is to be checked for voids, cracking in the adhesive, poor adhesive flow and porosity. Voids can be detected by tap hammer inspection but the repair should be also inspected with an appropriate ultrasonic technique.

Allowable defect sizes are specified in C5033 for defects in repair bondlines. These are based on the design overlap (L) and plastic zone length (l_p) for bonded repairs. The allowable defect limit is given by:

$$\phi_{max} = L - 1.5\Sigma l_p$$

Any defect smaller than ϕ_{max} will not result in the adhesive being the critical element in the repair. However, any defect in the taper region of a bonded repair requires the removal and replacement of the repair.

After repair of a region using elevated temperature curing adhesives or resins the region heated during cure needs to be inspected for blown core with an approved NDI technique.

5.2.3.6 Problematic Repair Procedures

A particularly helpful table lists the problems associated with bonded repair applications, how to find these problems and the consequences of the deficiencies. This table is included as Appendix D.

A procedure for the removal of defective bonded repairs is also described in C5033.

5.2.3.7 Prohibited and Conditional Repairs

Several repairs are prohibited under the C5033 standard unless experimental validation is carried out, including environmental testing. The repairs relevant to honeycomb panels are:

1. Injection repairs for face to core disbonds.
2. Potted repairs to sandwich-panels.

The standard points out that rather than fixing the problem, such repairs can lead to more severe damage as they can allow the ingress of moisture into the panel through the injection holes. This moisture can lead to panel damage such as corrosion at a later stage. If used, injection or potted repairs should be considered only a temporary repair method and for both techniques a patch must be bonded over the repair region for the repair to be considered permanent.

The standard specifically mentions defects that include traces of oil or fuel in the damage zone. It is stressed that the path by which the contaminants entered the panel must be found and repaired. The common practice is to simply dry the contaminant and proceed with the repair (this course of action is satisfactory only if the defect being repaired is also the source of contaminant entry).

5.3 Comparison of SRM and C5033

Engineering standard C5033 does not aim to be a repair manual and therefore the comparison with the F-111 SRM cannot be a direct one. Nevertheless both documents do contain repair details and the following comparisons addresses those areas where there is common ground.

5.3.1 Damage Assessment

It is somewhat difficult to compare the individual damage sizes permitted by the SRM and C5033, as they are described in quite different manners. Table 9 attempts to compare the guidelines between C5033 and the SRM.

Table 9. Comparison of allowable defects between SRM and C5033.

Damage Type	C5033	SRM
Face to core allowable void size	No dimension greater than 15 times thickness of the thinnest adherend and must not have a dimension >15% of the panel dimension.	Set size, depending on face thickness (see Table 5). The factor as used for C5033 ranges from 14 to 36 (for Repair Area A)
Distance between damage sites	As per Figure 24	Not specified
Total damage allowed on panel.	As per Section 5.2.2.2	As per Figure 19.
Defects at edges.	No defects within 0.125″ of any edge of a metal to metal bond.	Not specified.
Face to core de-bonds	0.5″ in diameter for critical structure	Allowable void length varies with face thickness from 0 to 1″ (repair area A).
Core splices	Maximum gap that is filled with adhesive is 0.125″ for critical structure.	Not specified
Core to core joints	Void must not exceed 0.5″ or 3 adjacent core cells in 12, or 5% of core to core bond area.	Not specified

The SRM does not have any restrictions on the distance between damaged areas or repairs. While the "p-factor" calculation (Figure 19) places restrictions on the total damage area of the panel, there are no guidelines on the acceptable distance between adjacent damage sites. In addition, the SRM fails to specify any minimum distance between a damage area and the edge of the panel and the implication is that repaired damage must be counted in the "p-factor" calculation (this may make many panels that otherwise would be repairable outside repairable limits). Similarly, specifications on core splices and core to core joints are not stated in the SRM, whereas these specifications are considered in C5033.

The damage size and location in limits in C5033 are taken from MIL-A-83376A. The use of these limits for metal-to-metal bonds is accepted, however, work is required to determine if they are applicable for face-to-core bonds.

5.3.2 Moisture Removal

Some reports [9,11,15] suggest that repair procedures involving elevated temperature cure of honeycomb panels can exacerbate the damaged areas by causing face to core disbonds due to a build up of pressure as the moisture turns to steam. This is related to inadequate moisture removal prior to cure of the adhesive. One article reported a face to core disbond occurring up to 1m away from the initial damage area due to inadequate moisture removal processes from the panel [9]. The SRM only specifies moisture removal adjacent to the damaged area, whereas, C5033 is more comprehensive in this respect as it specifies that if the whole panel is to be heated, then the whole panel must be adequately dried, not just the repair zone.

The use of vent holes to aid moisture removal in sandwich-panels is specified by the SRM, but this practice is not generally approved in the C5033 standard as it can lead to increased moisture ingress during subsequent operational service.

5.3.3 Surface Preparation

The surface preparation technique described in both the SRM and C5033 is similar, however three differences are noted as follows:

1. C5033 is more stringent in specifying the quality of the water that is used for the water break test and the aqueous silane solution. The SRM requires distilled water, whereas C5033 has details on the chemical content of the water permitted for use.
2. The time allowed to elapse between surface preparation steps is stricter in C5033 than in the SRM. C5033 has a range of allowable time gaps, ranging from 5 minutes to 1 hour depending on the atmosphere at the repair site, whereas the SRM has a set 30 minute allowed exposure time, regardless of the atmosphere. Also, while both standards emphasise the need for minimal exposure of a prepared surface, C5033 specifies that if a prepared surface is to be exposed for more than 5 minutes, it needs to be covered with an approved covering.
3. C5033 permits the use of corrosion-inhibiting primer, only if accurate control of thickness is attainable, and if there is proper containment and disposal procedures for toxic products. The application of primer is not covered in the SRM procedures for surface preparation.

Both the SRM and C5033 describe the same technique for the preparation of aluminium honeycomb core for bonding. Full surface preparation cannot be applied to the honeycomb due to its geometry and size and vapour degreasing and solvent washing are specified as acceptable alternatives.

5.3.4 Material Selection

The main difference between the two standards is that composite materials may be considered as a possible repair plate or patch material in C5033 for damage, whereas with the exception of the repair of stress corrosion cracks, only metallic repair plates can be bonded under the SRM specifications. Not all the materials listed in C5033 may be appropriate for a repair design to F-111. The materials in C5033 are those that have been approved for use in certain RAAF aircraft; however, the high operating temperature of some F-111 parts may make them unsuitable for use on that aircraft.

Table 10 compares the list of approved adhesives and resins in the SRM and C5033. There is no approved list of injectable potting compounds within C5033, which follows as injection repairs are prohibited under C5033.

Table 10. Approved adhesives and potting compounds for both SRM and C5033.

Material	C5033	SRM
Film adhesives for metal to metal and metal to honeycomb bonding	Cytec FM-73 Cytec FM-300 AF-130-2 AF-131-2 Hysol EA 9321	Cytec FM-73 Cytec FM-300 AF-130-2 AF-131 Hysol EA 9601 Plastilock 655
Core splice adhesives	Plastilock 654HE Plastilock 653 EX Cytec FM-404	Plastilock 654HE Plastilock 653 EX
Injectable potting compounds	none listed	Hysol EA 9317 NA A/B Magnabond Hysol EA934 NA A/B EA9321NA

5.3.5 Repair Design

The following table summarises the metallic repair plate specifications contained within each of the two standards.

Table 11. Metallic repair plate specifications for both standards.

Property	C5033	SRM
Material	Not specified	Same material as the damaged panel
Thickness	Thickness should be such that the stiffness of the patch and the panel match	Thickness is specified in each repair procedure – generally should be the thickness of the panel or greater
Tapering	Tapering of all patches, as per Figure 25.	Tapering required if patch is greater than 0.025″ thick.
Dimensions	Overlap lengths must be satisfied. Use specified equations to calculate required overlap lengths.	Overlap lengths must be satisfied. Required overlap lengths are specified depending on the face thickness.
Shape	Should be rectangular with curved corners.	No particular shape specified. (For circular patches, separate overlap lengths are specified.)
Surface profile	Surface profile needs to match that of the damaged panel.	Not specified

Although C5033 allows the use of alternative materials such as composites, for bonded plate or patch repairs, it also recommends that the material selections outlined in the SRM be used for repair. Some advantages have been shown for the use of composites to repair cracks in metallic structure [16], however for the types of repairs described in Section 5.1.3 there would be no advantage in using composite patches unless the panel had a high degree of curvature that would be more easily matched using composites.

The SRM and C5033 use different specifications for repair plate thickness, however, if one considers a metallic repair plate design, generally the result for both design specifications would be the same or similar repair plate thickness. Comparing Figure 22 and Figure 25 the main difference in the design of the taper for metallic patches or repair plates is the undercut that is specified in C5033 but not in the SRM. It has been showed that the undercut produces a lower peel and shear stress concentration at the edges of the patch [17].

C5033 provides a set of equations for the design of repair plate overlap and load transfer lengths rather than a look-up table based on structure thickness as detailed in the SRM. The resulting patch or plate lengths from the equations are similar to that produced by the SRM look-up-table. The methodology in C5033 is more

comprehensive and gives the engineer more understanding and control over the repair design process.

Finally when considering repair plate profile, if the repair plate does not match the surface profile of the damaged panel uneven bondlines that contain voids may result during application. This is particularly true for thick repair plates that will not readily form under the low pressures used during vacuum bag repair application. As mentioned above, this could be one time when moulding a composite patch to the surface profile could be an advantage over a metal plate, especially if the surface is curved in two directions.

5.3.6 Heating and Pressure Application

The literature [11, 15] is critical of the way heater blankets and thermocouples are used in some repair procedures and expresses uncertainty that the necessary adhesive cure cycles are being applied. This is mainly due to the presence of heat sinks in the structure and the positioning of thermocouples. Some SRMs only prescribe the use of one heater blanket for the whole repair area, regardless of substructure [11].

Both the F-111 SRM and C5033 have the same requirements for heater blanket use, including:

1. the repair region be divided into zones depending on the material, thickness and sub-structure
2. Each zone requires a separate heat source and monitor.

The main difference is that ceramic-backed heater lamps are expressly prohibited by C5033, however, the heater lamps to be used under the SRM are only described generically and no particular type is recommended or prohibited.

Both the SRM and C5033 prescribe the same location criteria for thermocouple positioning. The literature describes some haphazard positioning of thermocouples (for example placement 120° apart, near the repair area, regardless of substructure) and these procedures, while not prescribed by the F-111 SRM, are prohibited by C5033.

An important difference between the two standards is in the procedure for curing foam adhesives, as used in core splicing. C5033 does not approve the use of vacuum bags when curing these adhesives because it causes over-foaming of the adhesive and results in the formation of large voids. Hence a positive pressure should be applied when curing foaming adhesives. The procedures in the SRM indicate that vacuum bags could be used during the cure of core splice adhesives.

5.3.7 Post Repair NDI

C5033 appears to require more stringent post-repair NDI. C5033 specifies that a tap hammer test alone is not adequate for post repair inspection. The tap hammer test can be used in the first instance, but suspect areas should be further investigated using an ultrasonic technique. In comparison, in some repair procedures (D-12) the SRM states that a tap hammer test should be used for inspecting metal to metal bonds, with no further, more thorough investigations required. However, in other repair procedures (D-14, D-15, D-18), other NDI techniques are called up through another publication not reviewed in this report.

5.3.8 Prohibited and Conditional Repairs

A recent amendment to the F-111 SRM prohibits injection repairs (Repair D18) on the F-111 aircraft as a permanent repair. This was prompted by a large number of defects initiating at the site of previous injection repairs. Davis et al [9], describe the failure of the honeycomb at a previous repair site. The resin is injected into a void that has not been prepared for bonding, hence it is not possible to guarantee that the resulting bond would be durable. As there was previously no requirement for the repair to be covered (except at Deeper level or Operational level maintenance), moisture then enters through the injection holes and leads to corrosion and bond failure in the repaired region. This repair procedure is prohibited under C5033.

As discussed in Section 5.2.3.7, potted repairs to sandwich-panels are prohibited under C5033. C5033 highlights the fact that rather than repairing a problem, such repairs can lead to more severe damage, in the same way as an injection repair. The F-111 SRM does not prescribe any potted repairs. The repair procedures that require the removal of damaged face and core prescribe the installation of a core plug and cover plate.

As mentioned earlier, C5033 specifically discusses defects that are found to contain traces of oil or fuel. It is stressed that the path by which the contaminants entered the panel must be found and repaired. The presence of any contaminants, or consideration of the source of the moisture/contaminant, other than that found in the repair location, is not referred to in the SRM. C5033 also specifically prohibits the use of fasteners in conjunction with an adhesively bonded repair (except if the adhesive is the critical element in the repair).

5.4 Repair Concerns

Repair procedures or processes that exacerbate damage or defects in sandwich panels are the main concern. These can be placed into two categories as follows:

1. Poor moisture removal prior to repair leading to face-to-core bond failure during elevated temperature adhesive cure.

2. A repair process or procedure that increases the likelihood of moisture ingress that leads to degradation and damage later in the service life of the panel.

Both these issues are highlighted in C5033 but may not be adequately addressed in the SRM. Some work is required to modify the SRM to align with C5033 for the above.

6. Discussion

6.1 Honeycomb Panel Design

Honeycomb sandwich structures overcome the problem of increasing weight with increasing material thickness. The honeycomb core provides rigidity while preventing face buckling and keeping weight to a minimum. The core carries the shear load, while the faces take the compressive and tensile bending loads. The faces also resist the shear and normal loads applied in the plane of the fuselage skin.

For the F-111, the primary purpose of the panels is to provide shear support to the fuselage structure and provide an aerodynamic load path. The panels also withstand the internal loads generated by the structure. One of the major considerations in the original design of the panels was that the core should fail in shear prior to the faces de-bonding. This implies that the core shear strength is one of the determining factors for panel failure. Node bond degradation may effect the core shear strength and thus cause premature panel failure. Reduction of the core shear modulus and strength will tend to reduce the shear buckling, crimpling and face wrinkling resistance of sandwich panels. Conversely if the face-to-core bond strength degrades, a situation may arise where the face separates from the core prior to core failure by face wrinkling. Also, low adhesive flatwise-tension strength would be responsible for the face-to-core separations seen during repair operations.

Node bond and face-to-core bond strengths can degrade due to the presence of water inside the panel. It seems that degraded panel strengths from this cause were not considered during the original design. With the benefit of over 25 years of operational experience it is now obvious that RAAF aircraft are flying with panels in a degraded condition due to water ingress. Work is now planned to determine the significance of this type of degradation on panel strength and the criticality of primary panels to the overall structural integrity of the aircraft

6.2 Modelling

The basic FE model of the F-111 panel compared favourably with the OEM hand calculations used for initial design of the panels. This gave a reasonable indication that the modelling technique used was accurately reflecting the behaviour of the panels. It is hoped to use the modelling technique developed during the task to predict the

critical defect sizes for the various panels on the aircraft. Thus a validated, accurate modelling technique is vital. The initial analysis reported here, whilst not comprehensive, shows promise and more work is planned to validate the technique by comparison to experimental results.

The primary OEM load cases indicated that panel shear loads were caused by fuselage torsion and shear, the normal loads were caused by an aerodynamic pressure difference between the inner and outer surface and compression/tension loads were caused by fuselage bending. These load cases were applied to the model and the results showed that the panel face stresses were dominated by shear (approximately 70% of the stress field). The remainder of the stress field was shared between aerodynamic pressure (20%) and axial load (10%). The core shear stresses were similarly high with the flatwise tension loading in the core very low. The results also indicated a higher core shear stress near the edges of the panel.

This work implies that the main component load case that will lead to failure during service is shear in the core and face. This implies that degradation of the adhesive bond between the face and core (debonding of the face-to-core and failure of the fillet bond by shear) is of concern. Also, failure of the bond (or the presence of a disbond) between the face and core may cause face wrinkling of the face sheet leading to peel in the adhesive. The effect of degradation on adhesive peel strength thus may also be a concern. During repair operations it is likely that the flatwise tension strength of the face-to-core bond will be of more concern as the panel will be loaded by an internal pressure that pushes the face from the core. Additionally, moisture absorbed into the adhesive and the elevated temperatures used during cure of the repair adhesives may contribute to a lower adhesive strength during repair. Work is required to model this loading environment to better understand the effects of degradation and lower bond strength on the on the flatwise tension strength of the face-to-core bond.

The effect of gross face-to-core disbond was investigated using the model. In this analysis the core was assumed totally disbonded from the face. The results showed that the panel reached general instability at approximately five percent of ultimate load. More work is required to develop an understanding of the critical damage size for panel instability.

Issues that will need further consideration include:

a) The generation of a knock down factor for adhesive or core degradation in sandwich-panels that can be used in the analysis. The primary load cases, stress distributions and materials used in the F-111 panels need to be considered when developing this data.

b) The effect of panel curvature, and more realistic, non-uniform, panel load distribution.

c) Thermal and dynamic effects.

d) Repair operations leading to a build of internal pressure that may cause the face-to-core or fillet bond to fail through flatwise tension.

6.3 RAAF In-service Defects

The majority of the defects detailed in the reports associated with the two panels occurred at the panel corners and along the panel edges. This pattern of disbond damage was intuitively expected because the edge of the panel would be the most likely source of water ingress into the panel. The other defect sites were usually associated with an existing repair or some fitting that was attached to the panel where water may have entered through fastener holes or poor sealing. It is the presence of water that eventually leads to bond degradation, corrosion and debonding or voiding of the panel.

It is worthy of note that the results of the FE modelling indicate that the panel edges are subjected to the highest shear stresses where most of the defects are found in F-111 bonded panels.

6.4 Repair Procedures

The most notable issues raised by the review of SRM and C5033 repair procedures include the following:

1. The SRM sets allowable damage limits by considering the total percentage damage on the panel inclusive of any past damage (repaired or not). The calculation of the total percentage damage does not take into consideration the location of the previous damage on the panel. Also, as no credit is given to existing repairs on the panel (that is they must be considered as damage), eventually all panels on the aircraft if they sustain sufficient damage will be outside repairable limits. More work is required to examine the damage limits set in the SRM and C5033. It would be of benefit to the RAAF if a better understanding of how to set panel damage limits could be gained, particularly when considering the location and size of damage and if it is reasonable to give credit to repaired damage during damage size calculations.

2. C5033 has more detail than the SRM in the areas of:

   The area to be heated during repair application must be dried, not just the repair location, during moisture removal procedures. This is a concern as the cause of panel damage during maintenance is heat boiling moisture in the core, increasing the vapour pressure and causing the panel face to separate from the core. Complete drying of the entire panel or at least the total area that is to be heated during repair application is recommended.

Some heater systems that are known to be problematic and vacuum bag cure of foaming adhesives are banned. As above, during repair application heating of the panel is a major cause of panel damage. Any systems that may inadvertently cause overheating or damage to the panel would not be recommended.

Requires a higher level of post repair NDE specifying more than tap testing of repairs for acceptance. One of the routes for moisture entry into sandwich panels is through poorly applied or failed repairs. It is recommended that all steps are taken to ensure that a repair is properly applied and sealed from moisture and that no voids or disbonds exist that might allow a path for moisture.

C5033 allows for a wider scope in repair material selection, for example composite materials may be used for repair. It may be easier and simpler to form composite patches to highly curved surfaces and as such they would be a recommended alternative to metal bonded repairs in some cases.

7. Conclusions

1. Bonded sandwich-panels, such as those used for the fuselage of the F-111 aircraft, are generally designed to carry shear due to torsion and compression or tension due to bending.
2. Adhesive bond degradation will lower the face to core bond strength, fillet bond strength and core node strength. It is highly likely that this is the cause of premature face-to-core bond failure such as those seen in flight and during maintenance.
3. The F-111 panels are dominated by shear loading in both the faces and core with the highest shear loading at the panel edges. This has been confirmed by a preliminary Finite Element model of a generic panel. This model will be useful in further considerations of defect criticality.
4. The F-111 sandwich panels were designed such that the core shear strength would be lower than the face-to-core bond strength. Adhesive degradation may lower the face to core bond strength such that this bond fails before the core fails.
5. Node bond degradation may reduce the core shear strength so that sandwich-panels may fail prematurely by core failure.
6. The majority of defects found in RAAF F-111 panels are at the edges of the panel or adjacent to locations that may allow moisture ingress. Poor sealing, sealant failure or poor repair techniques are the main cause of moisture ingress into the panel. These issues need to be examined, as moisture is the main cause of panel/adhesive degradation.
7. Damage limits and the allowable spacing between damage on F-111 panels is poorly specified. Work is required to better understand these limits.
8. Some repair procedures are known to exacerbate damage on sandwich panels. It is important that these are identified and it is recommended that they be prohibited during maintenance operations.

9. The RAAF Engineering Standard C5033 is more comprehensive in its specification of processes and procedures for the design and application of bonded repairs than the F-111 SRM.
10. No consideration for adhesive degradation and associated reduction in panel strength was allowed for in the OEM design of the F-111 sandwich panels.

8. Future Work

This preliminary review has highlighted a number of problems associated with bonded sandwich panels on the F-111 aircraft in service with the RAAF. Work is now underway in a number of DSTO tasks to investigate these issues. With regard to the structural integrity of bonded sandwich panels, it is planned to develop a validated modelling technique for assessment and prediction of degradation and damage in such panels. Also, work will be undertaken to develop knockdown factors associated with adhesive degradation that can be applied to the design allowables for the panels. Finally, the modelling technique will be used to assess the damage limits and repairable limits as stated in the SRM for these panels, with and without degradation, to develop an understanding of how these limits were set and determine their level of conservatism. This will lead us to understand the effect adhesive degradation, panel damage size and location has on the structural integrity of bonded sandwich panels.

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Appendix A: Bonded Panel Construction

Table 12: Panel 3208 construction and material. The zones in this table refer to the areas indicated in Figure 16. [2]

Standard Panel Construction	Gauge and Material	Zones
Inner Face	0.120/2024-T81 0.071/2024-T81 0.032/2024-T81	1, 2, 7 3, 8 4, 5, 6, 9
Core	5052-1D 5052-1C 5052-1B	2, 3, 4, 5 7 to 9 6
Inner Face	0.020/2024-T81	2 to 6
Edge Member	0.012/2024-T81 Glass Fabric, 8-ply	1, 2 5, 7, 8, 9

Table 13: Panel 1102 construction and material. The zones in this table refer to the areas indicated in Figure 18. [2]

Standard Panel Construction	Gauge and Material	Zones
Outer Face	0.110/2024-T81 0.080/2024-T81 0.016/2024-T81	1 2 3
Core	P021-2B	2, 3
Inner Face	0. /2024-T81	3
Edge Member		1, 2

Appendix B: Defect Reports

This appendix contains the defect report numbers of the defect reports evaluated for panel 3208 in Section 4.2, and panel 1102 in section 4.3.

Panel 3208

Defect reports used for discussion:

	Defect number	Corrective Action	Corrective Action file
A	3AD/044/89	Repaired IAW SRA-F111C-345 D12 (oversize)	AIR1/4110/A8/1110D/2(38)
B	501WG/218/97	Issue of RFD/W F111-1293	501WG/AFENG/1293
C	501WG/217/97	Issue of RFD/W F111-1292	501WG/AFENG/1292 1 (5)
D	501WG/202/97	Issue of RFD/W F111-1241	501WG/AFENG/01241 Pt1 (5)
E	501WG/224/98	Issue of RFD/W F111-1487	501WG/AFENG/1487 Pt1 (3)

Other defect reports of damage described as "disbond", but with no further information are as follows:

Defect number
3AD/104/78
3AD/105/78
501WG/159/97

Defect report summaries containing no information on the defect nature or cause; or are irrelevant defects are as follow:

Defect number
3AD/070/82
3AD/067/83
3AD/061/83
3AD/023/84
3AD/022/84
3AD/126/86

Panel 1102

Defect reports used for discussion:

	Defect number	Corrective Action	Corrective Action file
A	3AD/021/90	Repaired IAW SRA-F-111-393	AIR1/4110/A8/1110D/3-
B	3AD/022/90	Sent to MRRL for repair action	AIR1/4110/A8/1110D/3-14
C	3AD/122/91	Sent to MRRL for repair action	AIR1/4110/A8/1110D/4/19
D	501WG/004/98	Repaired IAW AAP 7214.003-3 D14 (oversize)	-
E	501WG/041/94	Repaired IAW AAP 7214.003-3 D14 (oversize)	AIR1/4110/A8/1104d1 Pt1 (19)
F	501WG/051/98	Repaired IAW AAP 7214.003-3 D14 (oversize)	501WG/AFENG/1344 Pt1 (8)
G	501WG/103/94	Sent to MRRL for repair action	501/4110/A8/1110D Pt1 (8)
H	501WG/108/94	Repaired IAW AAP 7214.003-3 D14 (oversize)	501WG/4110/A8/1110D/1(4)
I	501WG/173/97	Repaired IAW AAP 7214.003-3 D14 (oversize)	-

Other defect reports of damage described as "disbond", or "corrosion", but with no further information are as follows:

Defect number
482SQ/263/84
501WG/124/94
501WG/206/96
501WG/216/97

Defect report summaries containing no information on the defect nature or cause; or are irrelevant defects are as follow:

Defect number
3AD/009/83
3AD/109/85
501WG/020/98
501WG/113/94
501WG/161/95
AUPFT/010/94
AUPFT/A05/94

Appendix C: Typical Manufacturing and Service Defects

This appendix contains tables of both the manufacturing and service defects found in honeycomb sandwich-panels, as specified in the RAAF C5033 Engineering Standard.

Table 14. Manufacturing defects in sandwich-panels. [3]

High Degradation (> 30%)	Medium Degradation (> 30% to 10%)	Low Degradation (<10% or enhanced)
1. Gap between core and edge member. 2. Voids in foam adhesive at edge members. 3. Mismatched nodes in corrugated core. 4. Incomplete edge seal.	1. Unbonded nodes. 2. Gaps at machined face steps and core. 3. Crushed core at edge members. 4. Blown Core. 5. Over-expanded core.	1. Core splice separation. 2. Diagonal line of collapsed cells. 3. Drilled vent holes in face. 4. Sideways condensed core. 5. Incomplete core splice. 6. Misaligned ribbon.

Table 15. Service defects in metallic sandwich-panels. [3]

Service Defects	Significance	Comments
Debonded edge members.	High. All load transfer into sub-structure occurs at edge member.	Usually associated with poor surface preparation during manufacture. Injection repair useless. Remove edge member, prepare correctly, re-bond.
Corroded core.	High. Face instability and poor shear strength.	Usually associated with panel defect or past repair. Injection repair useless. Fix panel defect as well as corrosion.
Debonds between face and core.	High. May lead to instability of face, lack of shear integrity.	Usually associated with impact damage, poor surface preparation or inclusions in bond line. Injection repair useless. Remove disbond, damage area and apply repair.
Bond line inclusions.	Moderate. Usually a manufacturing defect, often found in service.	Injection repair useless. Remove material and inclusion and apply repair. External patch may be possible.
Dented faces.	Moderate. Large dents; instability at high loads.	Small dents require only aero smoothing. Injection repair may lead to corrosion. Remove face in damaged area and apply repair to core and face.
Cracked edge members.	High. Load transfer to sub-structure is through edge members.	Remove edge member and replace. Bonded patch may be used to repair.
Corroded Face.	Low. Face stresses are usually very low.	Remove damaged face and apply repair.
Face penetration.	Moderate. Face stresses are usually very low.	Repair to restore environmental protection. Remove damaged face and core and apply repair.
Surface scratches.	Low. Stress levels are low, no fatigue.	Remove sharp edges, restore corrosion protection.
Moisture entrapment.	High. Will lead to corroded core.	Usually associated with panel damage, old repair or debond. Repair **cause** to prevent further damage. Do not simply dry.
Fuel, Oil Entrapment.	High. Indicates debond or panel damage.	Repair **cause** to prevent further damage. Do not simply dry. Injection repair useless.
Fatigue.	Low. Stresses in bonded panels are low.	Indicative of other damage which should be repaired as well as cracking.
Stress Corrosion.	Low in faces. High in edge members .	Doesn't occur in faces. Repair edge members by replacement or bonded repair.

Appendix D: Bonded Repair Processing Problems

Table 16. Defects in adhesive bonds caused by processing errors. [3]

Processing Problem	Effect/Comments	Defect	Significance
Incorrect vacuum bag formation.	Poor pressurisation.	Voids, porosity.	Moderate.
Incorrect heating procedures.	Under-cure of adhesive: • Check penetration resistance with 2H pencil. Overheating of structure: • Metallurgical damage. • Delaminations in composites. Overheating of the adhesive: • May result in charring, crazing, discolouration. • Prolonged heating at correct temperature has no effect.	Overheating or undercure of adhesive. Damage to structure.	High.
Incorrect heat-up rate.	Too rapidly traps volatile materials resulting in micro-voids. Too slowly causes poor wetting of surface.	Porosity. Poor adhesion.	Moderate to High.
Moisture evolution during heating due to: • Inadequate drying. • Contaminated adhesive. • Incorrect environment.	May occur in composite and honeycomb materials. Small amounts cause voids due to localised gas formation. Large amounts generate high pressures which may delaminate or debond the panel.	Voids, porosity, debonds, blown core.	High.
Inadequate surface preparation.	Large area debonds at the interface in service. Durability reduced. Not detectable at manufacture by lap shear test, tap hammer, NDI. A service defect, although its cause occurs in production. Injection "repair" is useless.	Long term bond failure. Corrosion under joint.	High.
Incorrect mixing of two part adhesives.	Uneven or incomplete cure.	Low bond strength.	High.
Adhesive out of shelf life.	Low flow, poor wetting of surface.	Low bond strength and durability.	High.

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19. ABSTRACT

Many of the fixed and removable panels on the RAAF F-111 aircraft are made up of bonded honeycomb sandwich panels. Experience with the RAAF fleet has shown that a serious problem exists with degradation and damage of these panels. A review of the literature was undertaken to gain an understanding of the extent of this problem. It was found that panels were subject to large areas of adhesive bond separation and corrosion damage. This damage was believed to be caused by the ingress of water in the panel through poor sealing at the edges or after repair of the panels. Moisture in the panel is also believed to cause adhesive degradation that may reduce the strength of the bonds in such panels. At the same time the literature was surveyed to determine the design load cases for such panels. This information was used to develop a simple finite element model of a bonded honeycomb sandwich panel. This model was in turn used to generate data on the loading and failure of such panels. In addition, an understanding of current repair techniques was gained by review of the F-111 Structural Repair Manuals and the RAAF Engineering Standard C5033.